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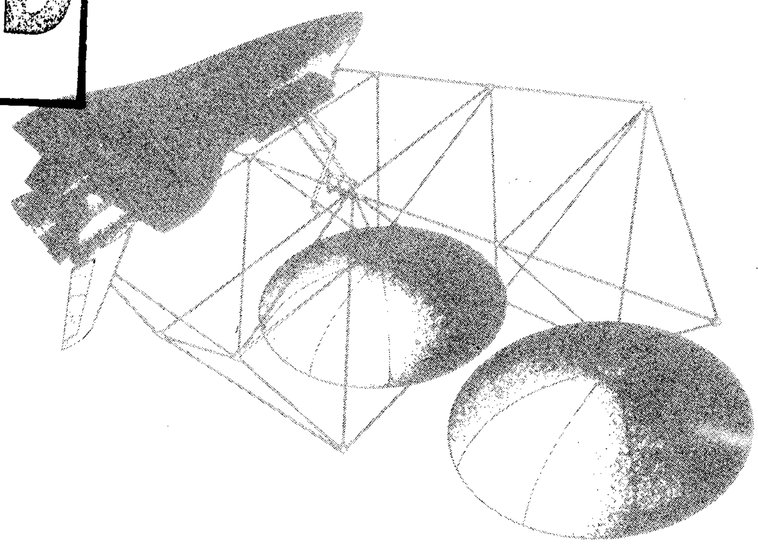

Large Space Systems Technology

An Industry/Government Seminar
held at NASA Langley Research Center
Hampton, Virginia, January 17-19, 1978

NASA

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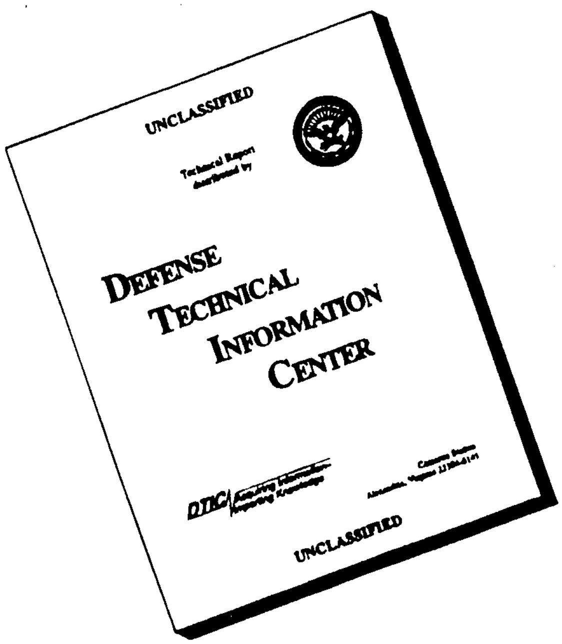

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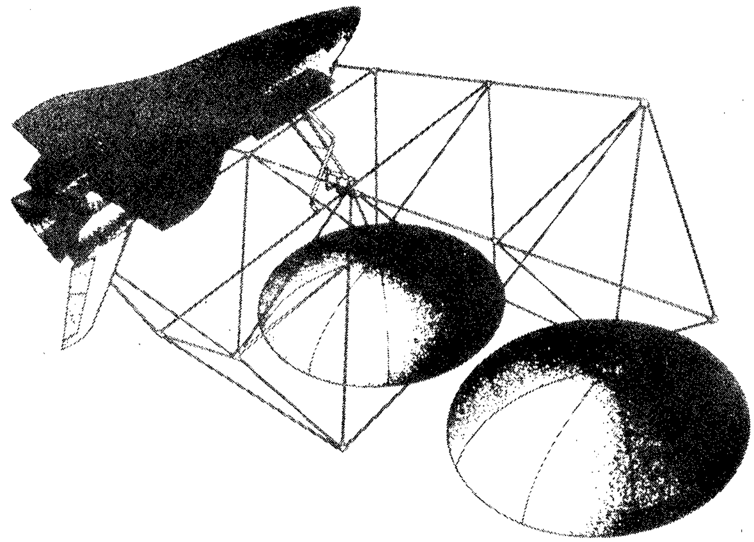

Large Space Systems Technology

Compiled by
E. C. Naumann
Langley Research Center
and
A. Butterfield
General Electric Company

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NASA
National Aeronautics
and Space Administration

Scientific and Technical
Information Office

1978

PREFACE

These Proceedings include a compilation of the papers presented during the Seminar and a summarization of pertinent questions and answers arising during the presentations. The Seminar provided Government and Industry representatives with an opportunity to exchange information, to review the present status of related technology, and to plan the development of new technology for Large Space Systems. The Seminar began with four invited papers: three NASA papers to provide industry and the user community with information pertinent to NASA's planning and forecasting efforts, and one Industry paper to solicit for NASA, by way of a questionnaire to industry, information that would help NASA in planning the total program.

Selected papers, solicited from Government and industry, describing items of technology or developmental efforts followed the invited papers. These papers were divided generally into two major areas of interest: the first group addressed subjects pertinent to large antenna systems; the second group addressed technology related to large space platform systems. The Seminar concluded with a Forum and Issues session with participants from both Industry and Government.

This compilation provides the participants and their organizations, in a referenceable format, the papers presented at the Seminar. The Large Space Systems Technology Program Office, Langley Research Center, which sponsored the Seminar, will utilize this information as an aid in their planning and in the definition of technology goals and technology developments.

At NASA LaRC, the work was monitored by E. C. Naumann, Task Manager and Seminar Coordinator. A. Guastaferro, Manager of the LSST Program Office served as General Chairman of the Seminar. The Task was administered by S. M. Scala, Senior Consulting Scientist, General Electric Company, who also served as Moderator of the Forum/Issues Panel.

CONTENTS

VOLUME II

ACRONYMS AND ABBREVIATIONS

The list of acronyms and abbreviations contains both commonly used terms and specific items not immediately defined within a presentation. Single usage acronyms readily defined within the text, U.S. Government agencies, and spacecraft acronyms do not appear. In addition, the technology for Large Space Systems has begun to generate descriptive words coined or extracted from common usage terms; such words with obvious meaning have not been included.

ACS	Altitude Control System
AMB	Amplitude Modulated Beam
AMVSB	Amplitude Modulated Vestigial Side Band
ATS -(6)	Advanced Technology Satellites (No.)
AWG	American Wire Gage
BER	Bit Error Rate
BW	Bandwidth or Beamwidth
CCD	Charge Coupled Devices
CCIR	International Radio Consultative Committee
Cervet	Trade Name for a Glass Ceramic with a very low Coefficient of Thermal Expansion
C.G.	Center of Gravity
CMG	Control Moment Gyro
CMD	Command
CONUS	Continental United States
CoTE	Coefficient of Thermal Expansion
CRT	Cathode Ray Tube Display
CW	Continuous Wave

Cy, CY	Calendar Year (Jan. 1 thru Dec. 31, inc.)
DCP	Data Collection Platforms
DDR&E	Department of Defense, Research and Engineering
DOF	Degrees of Freedom
DSN	Deep Space Network
d/t	Ratio of Diameter to Wall Thickness for a Tube
EM	Electro-Magnetic
ERP	Effective Radiated Power
E	Extensional stiffness
EVA	Extra Vehicular Activity
f, F	Frequency
FUV	Far Ultra Violet Light
Fy, FY	Fiscal Year (now Oct. 1 through Sept. 30, inclusive)
g, G	Gravity
GCP	Geostationary Communication Platform
GEO	Geosynchronous Earth Orbit
GG	Gravity Gradient
GPC	General Purpose Computer
GRC	General Research Corp.
GR/E	Graphite Epoxy
(GY--)	(Graphite Epoxy Formulation Code)

GTD	Geometric Theory of Diffraction
GTP (GR/TP)	Graphite Thermoplastic
h	Film Transfer Coefficient for Thermal Conductivity
HEO	High Earth Orbit
HMS	Graphite Fiber Trade Mark, High Modulus Series
IC	Integrated Circuit (Microcircuit)
ICD	Interface Control Document
IF	Intermediate Frequency
INC	International Nickel Co.
IR	Infra Red
IR&D	Independent Research and Development
IRU	Inertial Reference Unit
ISP	Specific Impulse
IUS	Interim Upper Stage
L	Altitude in Equivalent Earth Radii
LED	Light Emitting Diode
LEO	Low Earth Orbit
LEPS	Laser Electric Propulsion System
LSS	Large Space Systems
LSST (LASST)	Large (Area) Space System Technology
(ATLASS)	(Advanced Technology for LASS)

LTL	Low Thrust Liquids
MCDS	Manual Control Display System
MCDU	Manual Control Interface Unit
MDM	Multiplexer-Demultiplexer
MF	Mass Fraction (Percent of total weight carried as fuel)
MFG	Manufactured
MHD	Magnet Hydro Dynamic
MMH	Mono methyl Hydrazine
MOUSE	Computer Program for Modal Optimization and Analytical Model Updating (Test data)
NASCAP	Computer Program for Analyses of Electrical Charges
Nd/YAG	Neodymium doped Yttrium Aluminum Garnet
NE ΔT	Noise Equivalent Temperature
NF	No Failure
NUV	Near Ultra Violet
OFT	Orbital Flight Test
OMS	Orbital Maneuvering System
OOA	On-Orbit Assembly
OTV	Orbit Transfer Vehicle
P	Protons

P, $Pcr_{(exp, th)}$	Compression Load applied to a column, cr, loading which causes buckling (experimental, theoretical)
Paramp	Parametric Amplifier
POP	Program Operating Plan
RBV	Return Beam Vidicon
R&D	Research and Development
R&T	Research and Technology
RDT & E	Research Development Test and Engineering
RCS	Reaction Control System
rf RF	Radio frequency
Re	Earth Radius
RFP	Request for Proposal
RMS, rms	Root Mean Square for surface accuracy or statistical error
RMS	Remote Manipulator System
RT	Room Temperature
RTG	Radio isotope Thermoelectric Generator
RTOP	Research and Technology Objectives and Plans
RTR	Research Technology Resume
RTV	Trade name for silicone elastomeric materials
SAR	Synthetic Aperture Radar
SEPS	Solar Electric Propulsion Stage
SL	Side Lobe

S/N	Signal to Noise Ratio
SPAR	SPAR Aerospace Products Ltd.
SPAR	Computer Program for General Purpose Structural Analysis
SPS (SSPS)	Solar Power Satellite (Space SPS)
STS	Space Transportation System
TASO	Television Allocation Study Organization
TDRSS	Tracking and Data Relay Satellite System
Tg	Glass Transition Temperature
TR-SW	Transmit-Receive Switch
TV	Television
TVBS	Television Broadcast Satellite
T/W	Thrust to Weight Ratio
UV	Ultra Violet
UDT --	Electronic Part Numbers for United Technology Devices
ULE	Trade name for a Glass Ceramic with a very low coefficient of Thermal Expansion
VCM	Vapor Condensable Material
VSB	Graphite Fiber Trademark

ABBREVIATIONS FOR UNITS AND MEASUREMENTS

The following symbols and abbreviations have been used within this compilation.

Å	Angstrom units, 10^{-10} meter
A, amp	amperes
AU	Astronomical Unit
arc min	angular measurement in minutes of a degree
arc sec	angular measurement in seconds of a degree
b/sec, B/SEC	bits per second
BTU	British thermal unit
C, °C	Celsius degrees
cm, CM	centimeters
db, dB, DB	decibels
dbw, DBW	decibels relative to one watt
eV, EV	electron volts
F, °F	Fahrenheit degrees
ft., FT, '	feet
G -	giga (X 10^9) multiplier
GHz	gigahertz
GW	gigawatt
GWe	gigawatt electrical

gm, Gm	grams
g/cm^2	area density, grams per square centimeter
Hz	hertz
in, IN, "	inches
°K, K	degrees Kelvin
k-K	kilo ($X\ 10^3$) multiplier
KA	kiloampere
Kb/s, KB/S KBPS	kilobits per second
kev, KEV	kilo electron volts
kg, KG	kilogram
km, KM	kilometer
KMC	kilomegacycles (same as GHz)
kv, KV	kilovolt
kw, KW	kilowatt
kw-hr	kilowatt hours
Kwe, KWE	kilowatt electrical
lbs, LBS	pounds
lb-ft	pound feet (torque)
m, M	meters

M -	mega, million (10^6) multiplier
M bits/sec	megabits per second
MHz	megahertz
MW	megawatt
MWe	megawatt electrical
meV, MEV	million electron volts
Micron	10^{-6} meter
MI	mile
Mil	10^{-3} inch
m-- M--	Milli -- (10^{-3}) multiplier
mm, MM	Millimeter 10^{-3} meters
ms	milliseconds
nm, NM	nautical miles
N-m, N-M	Newton meters (torque)
n-m-s, N-M-S	angular momentum, Newton-meters-seconds
psi, PSI	pounds per square inch
rad, RAD	radian
RAD/SEC	radians of angle per second
ST MI	Statute mile
TORR	pressure, equivalent to millimeters of mercury
V	Volts
W	watts

λ	wavelength
μ	micro (10^{-6}) multiplier
μm	10^{-6} meter
$\Downarrow + \Uparrow$	perform the operation in two opposing orientations relative to the force of gravity
#	Number of

CONVERSION FACTORS FOR UNITS

U.S. Customary Units	SI Units	Multiply by
ampere (International)	ampere	0.9998
BTU	joule	1055
electron volt	joule	1.6×10^{-19}
Fahrenheit (temperature)	Kelvin	$5/9\ (t_F + 459.67)$
Fahrenheit (temperature)	Celsius	$5/9\ (T_f - 32)$
foot	meter	0.3048
inch	meter	0.0254
pound force	newton	4.448
pound mass	kilogram	0.4536
mile	meter	1.609×10^3
nautical mile	meter	1.852×10^3
slug	kilogram	14.594
psi	pascals	6.895×10^3

INTRODUCTION

The availability of the Space Shuttle transportation system will make it possible to deploy, erect and/or eventually fabricate on orbit large space systems (LSS) beginning in the decade of the 1980's. Preliminary studies conducted by NASA, DOD, and the Aerospace Industry indicate that in order to meet future user needs, large antennas and platforms will be required either in low Earth orbit or in geo-synchronous orbit. Specific applications have been identified in a series of recent studies which have examined and evaluated future civilian and military space possibilities with relevance to human and/or defense needs as the basic measures.

One of the most comprehensive recent studies resulted in two NASA reports, issued in January 1976: "Outlook for Space" (NASA SP-386), which identified potential future space activities, and "A Forecast of Space Technology" (NASA SP-387) which provided a comprehensive forecast of technology which might reasonably be expected to be available for the management of information, energy, and matter in space during the last two decades of the 20th century.

A three-day Industry Workshop on Large Space Structures was held at the NASA Langley Research Center in February 1976, to help NASA identify the technology developments required for these proposed missions. At this workshop, representatives of several Aerospace Companies were asked to respond to a Key Issue Questionnaire. These responses were published in two NASA reports: A Compilation of Company Presentations (NASA CR-144997); and An Executive Summary (NASA CR-2709).

In March 1977, the Langley Research Center was named lead Center of a multicenter, multidisciplined planning activity with the mission of defining and developing critical technology for use in large space systems in the years 1985 to 2000. The Large Space Systems Technology (LSST) Program Office evolved from these planning and program definition activities. This seminar was sponsored by the LSST Program Office to provide a forum for the more effective interchange of ideas, plans, and program information needed to develop the required large space system technology. The format of the Seminar is closely aligned to that of the 1976 workshop because of the effective interchange obtained during the workshop.

The seminar organizing committee utilized invited papers, contributed papers, and panel discussions to maximize potential benefits for each of the participating organizations. The invited papers were used to provide industry an insight on the views of NASA Headquarters, the LSST Program Office, and background information on Shuttle astronaut interfaces, and to provide NASA information on Industry views by means of a questionnaire. The contributed presentations were essentially equally divided between industry and Government. These papers emphasized on-going or planned in-house technology development in support of large antenna systems or large platform systems. Typical subject matter

addressed at least one of the following: mission requirements, structural concepts, materials, controls, structural alignment, thermal control, metrology, and packaging/shuttle interface considerations. Finally, the last session of the Seminar was devoted to a Forum, the purpose of which was to provide industry and government representatives with the opportunity to present their views on significant and/or controversial issues, to answer questions from the attendees, and to focus attention on critical LSST needs and approaches.

The proceedings of the Seminar are presented in this report and in an Executive Summary (NASA CR-2964). This report contains all the invited and contributed formal presentations given during the Seminar and also includes those contributed papers which were considered to be informative and useful but which could not be delivered during the program due to time constraints. The Executive Summary condenses and records the findings and conclusions of all the presentations and, in addition, highlights the comments and recommendations of the members of the Forum Panel.

EQUIPMENT INSTALLATION

ON

LARGE AREA SPACE SYSTEMS

E. KATZ

CHART 1 - EQUIPMENT INSTALLATION ON LARGE AREA SPACE SYSTEMS

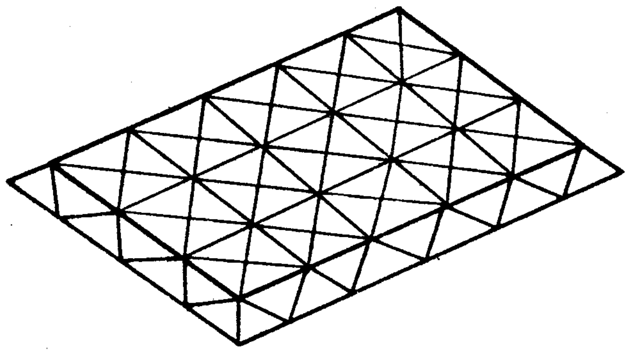
This paper is concerned with the requirements and concepts for the installation of various types of mission and subsystem equipment on large area space systems. The paper is a synthesis of work performed under company discretionary programs, contractual studies with the Langley Research Center, and current year IR&D plans.

CHART 2 - TYPES OF STRUCTURAL PLATFORMS

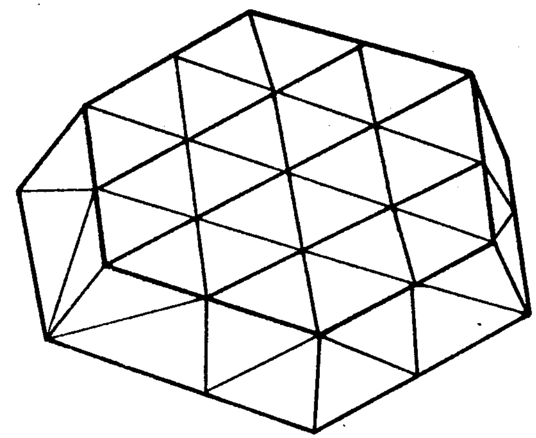
Equipment installations will be required on various types of large structural platforms. The major exception to this requirement is the case of fully deployable systems or sub-assemblies, where the equipment is preinstalled within the packaged system. In most other cases, the equipment will be installed in a series of module placement and interconnect operations. This is because the large area of the platforms will dictate that some, if not most, of the modules and their functions be located at points distant from one another.

TYPES OF STRUCTURAL PLATFORMS

DEPLOYABLE PLATFORM



ERECTABLE PLATFORM



FAB-IN-SPACE PLATFORM

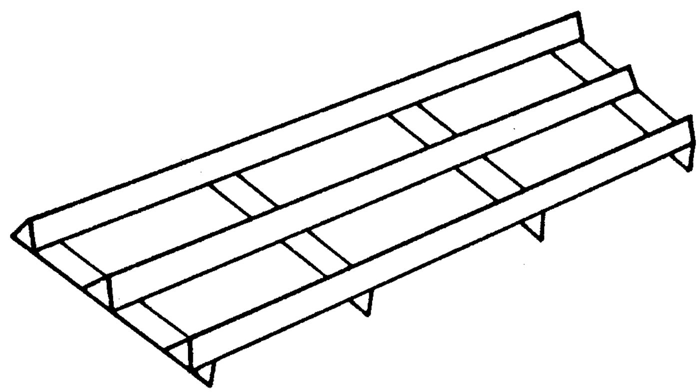


Chart 2

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Space Division

CHART 3 - SCOPE

In the material which follows, two conditions are assumed which scope the applicability of the present conclusions.

First, the concepts are limited to the earliest presumed space constructions; i.e., the mid-80's time period. In this context, it is assumed that the system sizes, designs, construction techniques and equipment, and operations will be somewhat less advanced (although efficient for the jobs at hand) than would be expected for a later time period.

Second, it is assumed that all construction operations in the earliest time period would be performed by the Shuttle system. Previous studies have attested to the validity of this assumption.

SCOPE

- SHUTTLE-BASED ASSEMBLY OPERATIONS
- EARLY SPACE CONSTRUCTIONS

Chart 3

CHART 4 - TYPES OF EQUIPMENT

The purpose of this chart is to illustrate the wide range of weights, sizes, and functions which the installed modules may exhibit. By no means are all the likely types of equipment illustrated--but those shown are a representative sample. It may be noted that several of those illustrated will require unique locations on the platform (e.g., the ion thrusters which could be required for stationkeeping). Although not illustrated, one of the most challenging installations will be the interconnecting power and signal cable sets about which more will be said later.

TYPES OF EQUIPMENT

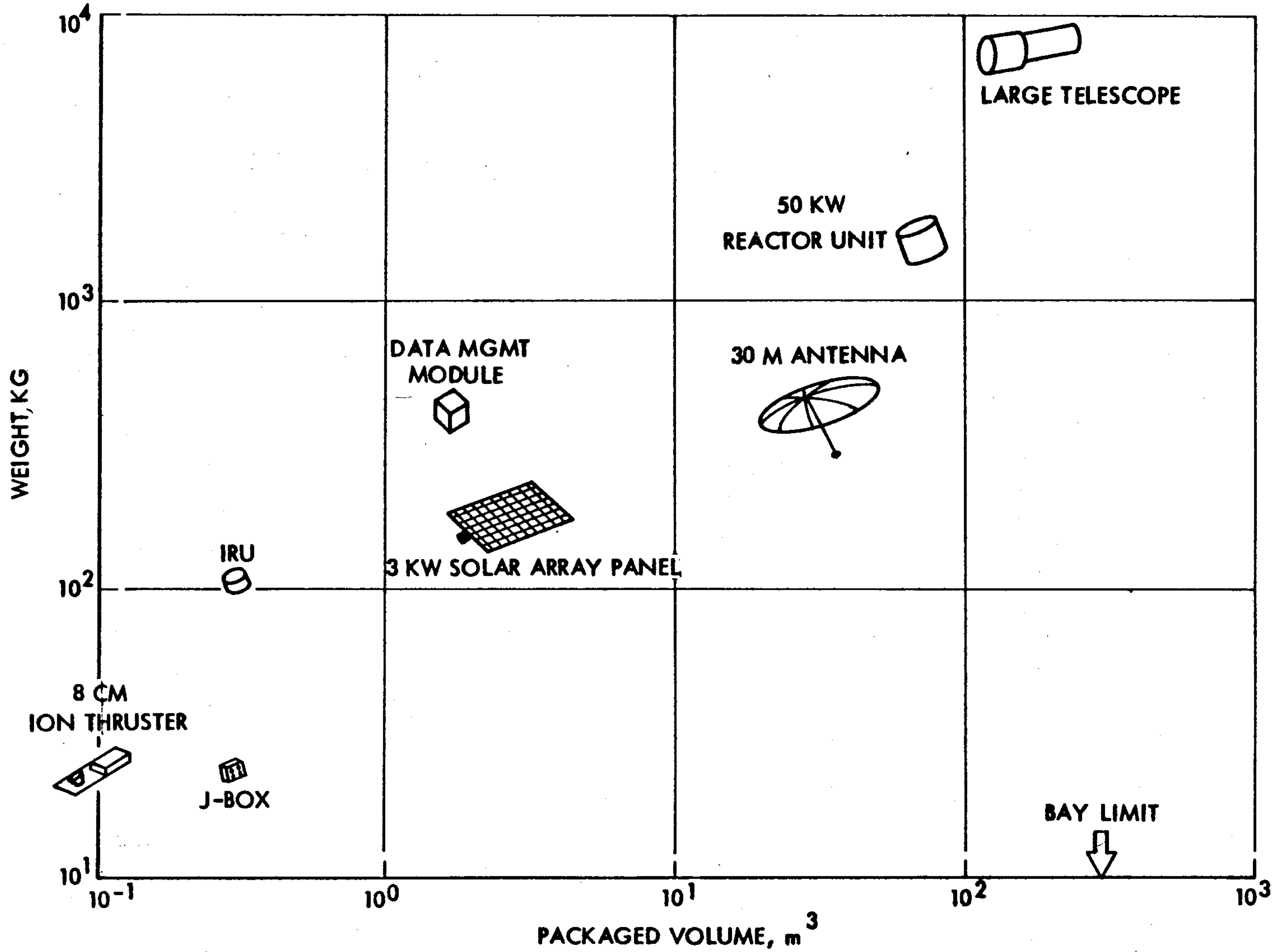


Chart 4

CHART 5 - EQUIPMENT INSTALLATION INTERFACES

The general requirements for the equipment installation are that it be mechanically tied to the structure and electrically interconnected with power and signal sources/receivers. More specifically, those interfaces must be "easy" to make and verify in the orbital environment. The electrical interface is shown to be separate from the mechanical; although integral mechanical-electrical interfaces are possible, they may not be most appropriate for the earliest constructions. The module should generally be installed at a structural node ("hardpoint") where loads can be most effectively distributed. As shown in the following chart, a single node attachment is generally sufficient to accept all but the very largest dynamic modules.

EQUIPMENT INSTALLATION INTERFACES

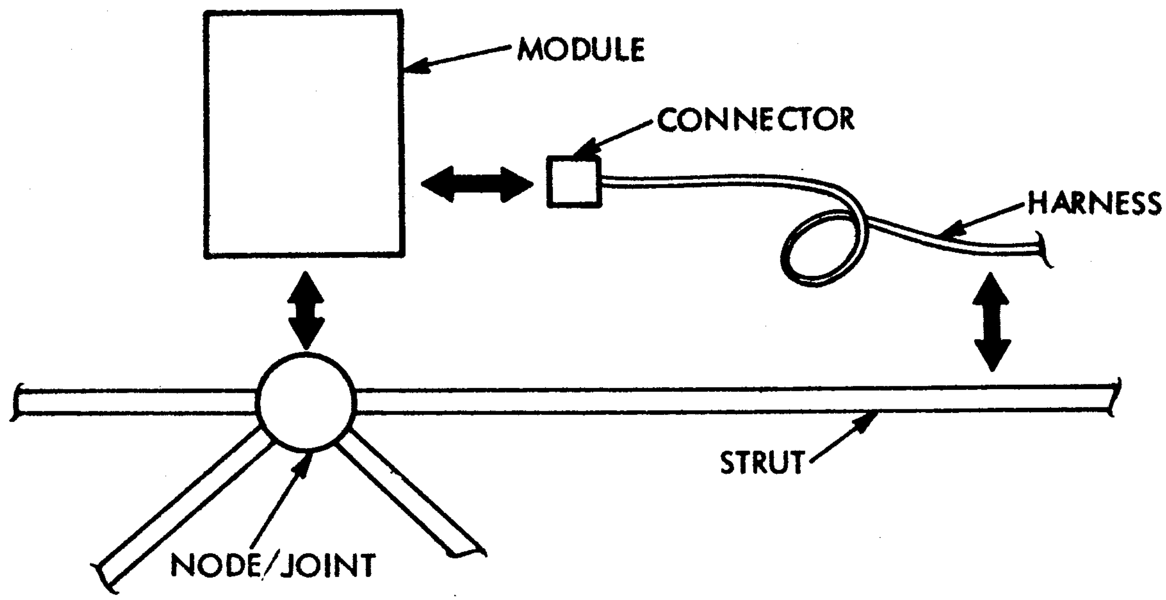


Chart 5

CHART 6 - INTERFACE LOADS

This chart shows the moment expected at the mechanical interface of a "representative" module ($\approx$1 ton, 10-ft height). For this case it is assumed that the module produces no dynamic loads, and that the loads at the interface are those required to react the attitude excursions of the platform system and thrusting during orbit transfer. The graph is plotted as a function of the control frequency to represent the effect of "stiffness" in the control loop. As noted, the interface moment is fairly modest and, at the control frequencies and thrust-weight ratios of interest, is well within the capacity of a lightweight structural node/joint to react. If, of course, the platform were to require fast-slew operations or if--as in the case of a chemical thruster--dynamic loads were to be induced, a multi-node interface could be required.

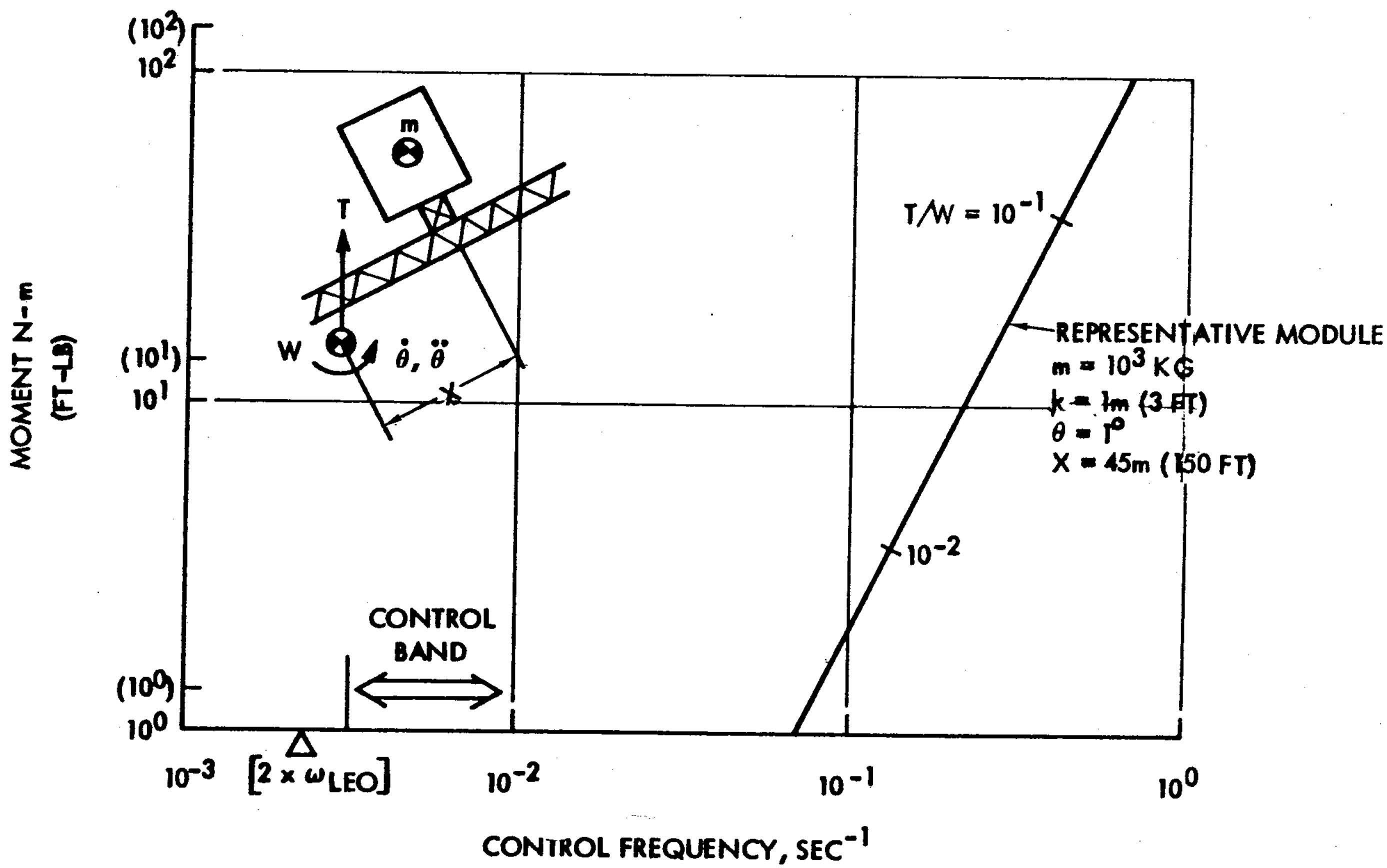

MOMENT N-m
(FT-LB)
(10^2)
10^2
(10^1)
10^1
(10^0)
10^0
10^{-3}
[$2 \times \omega_{LEO}$]
10^{-2}
10^{-1}
10^0
CONTROL FREQUENCY, SEC^{-1}
CONTROL
BAND
T
W
m
$\dot{\theta}, \ddot{\theta}$
$T/W = 10^{-1}$
10^{-2}
REPRESENTATIVE MODULE
$m = 10^3$ KG
$k = 1$m (3 FT)
$\theta = 1°$
X = 45m (150 FT)

Chart 6

CHART 7 - MECHANICAL CONCEPT

A concept of a mechanical interface/adapter is illustrated. This is essentially a probe-drogue type of coupling where the probe is a part of the module. The drogue is incorporated within the multi-strut union as shown in overall view in the upper right-hand diagram. In addition to the interface requirements stated earlier, the coupling must allow an accurate positioning of the module with respect to the node, must react loads and must provide a rigid base for the module. The probe (shown here to be about 0.3-m/1-ft length) is engaged in a two-step latching procedure. As later discussed, the module will be positioned "over" the node and slowly "lowered" into position. In the first step, the ball end of the probe will engage the drogue and its latches. In the second step, the probe's redundant motors will drive the tapered shell of the probe into a "jam" with the walls of the drogue. The motors would be torque-limited and reversible so that disengagement could be effected, if warranted. Power for the motors will be discussed later.

MECHANICAL CONCEPT

- ACCURACY
- LOADS
- DYNAMICS

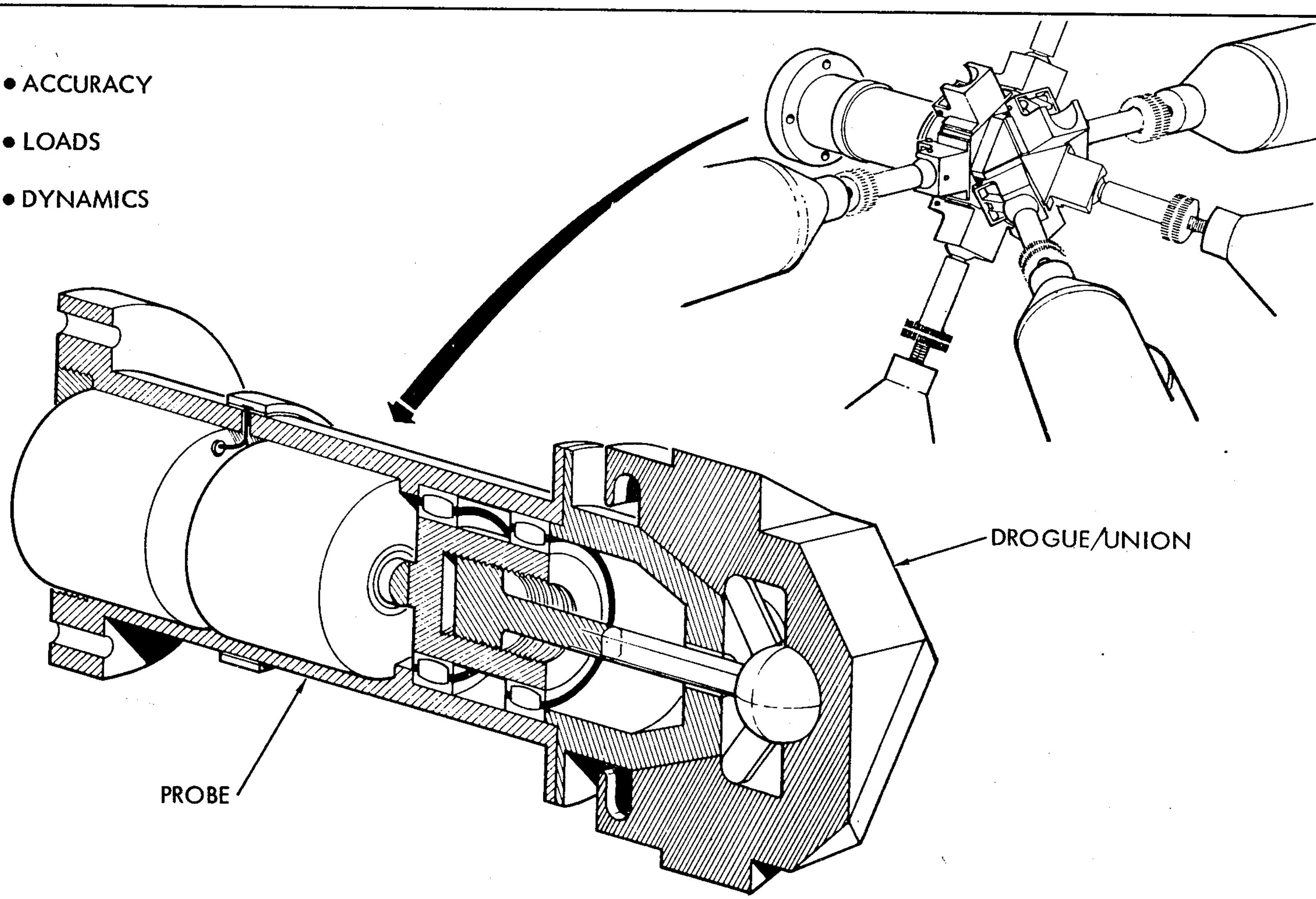


Chart 7

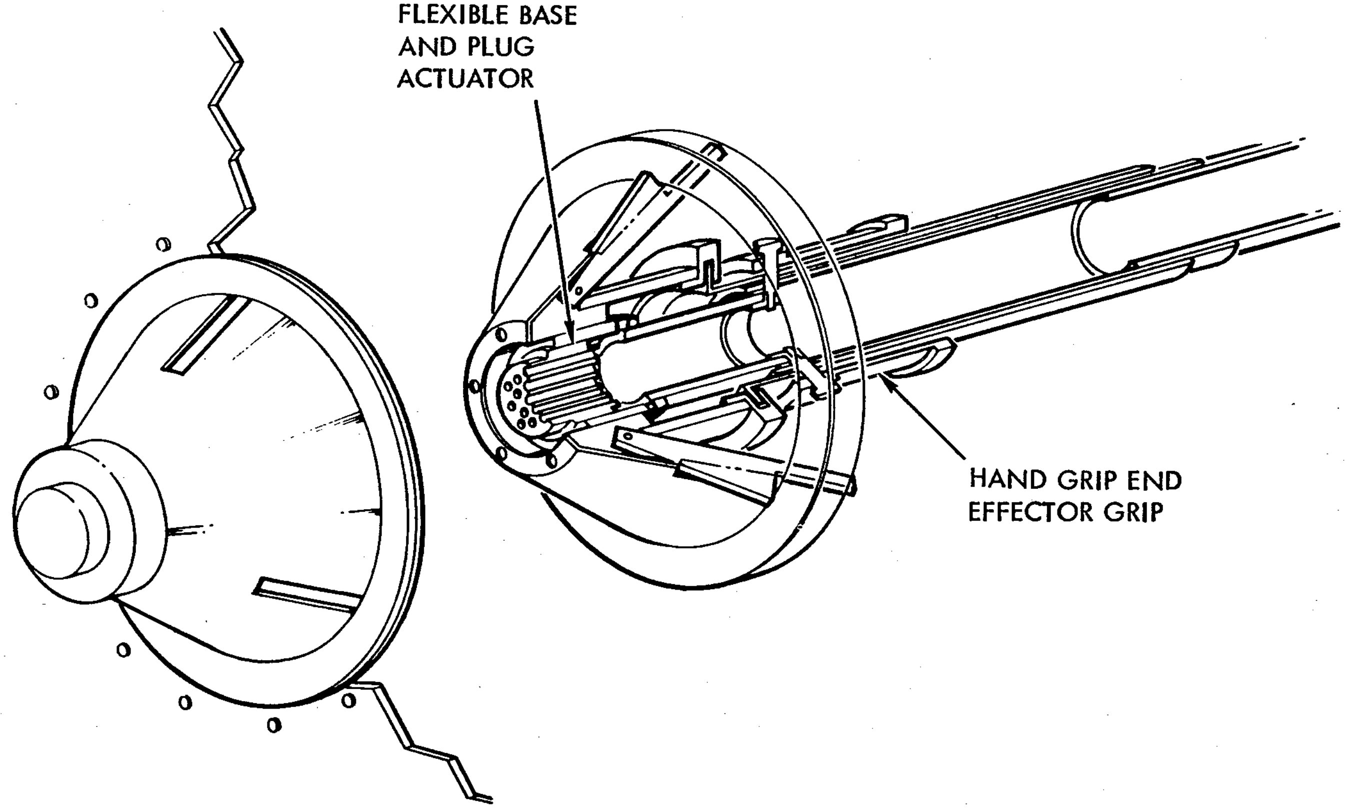
CHART 8 - ELECTRICAL CONNECTOR ASSEMBLY

A concept of an electrical interface/connector is illustrated. As in the previous mechanical device, this is also a probe-drogue type of coupling. The probe is part of the harness assembly and contains the male connector, and the drogue is built into the sidewall of the module. In addition to the earlier stated requirements, particular attention must be given to alignment (to avoid pin-bending) and to the force requirements. Since this concept is envisaged to be compatible with EVA or with manipulator arm operations, these factors are of prime importance. In addition, there should be some way in which the completed connection can be verified and in which the connection could be disengaged, if required.

As in the mechanical concept, this coupling is made in a two-step operation. In the first step the probe is gripped by the astronaut or effector on the outer sleeve and the unit inserted into the drogue. To permit initial misalignments, the probe is designed to have a compliance with respect to the harness/connector. When the probe is fully inserted, its spring-loaded latches will engage the drogue's slotted shell. In the second step, the astronaut/effector will "pull" on the probe's outer sleeve to extend the male connector into a mate with the female in the drogue. The action is designed with a 7:1 mechanical advantage and with key-ways to assure precision pin engagement.

ELECTRICAL CONNECTOR ASSEMBLY

- ALIGNMENT
- CONTINUITY



- FORCE
- DISCONNECT

Chart 8

CHART 9 - INSTALLATION TECHNIQUE

This chart illustrates one possible concept for the installation of modules on the structure. In this case, the Orbiter is shown equipped with two RMS (Remote Manipulator System) arms. The forward arm maintains the Orbiter's location with respect to the structure and also provides TV viewing of the module placement operations. The second arm has grasped the module at the probe end and executes the detailed installation operations. Power to drive the probe motors would be provided through the arm/end-effector to brush pickups on the probe. These operations will, of course, be under the control of the crew at the aft deck control station, and adequate lighting and direct- and TV-viewing will be essential. When the mechanical and electrical connections are completed, the two arms may be used to "walk" the Orbiter to the next installation location--as shown in the following chart. The potential of the RMS arms to perform these operations will be discussed in a later chart.

INSTALLATION TECHNIQUE

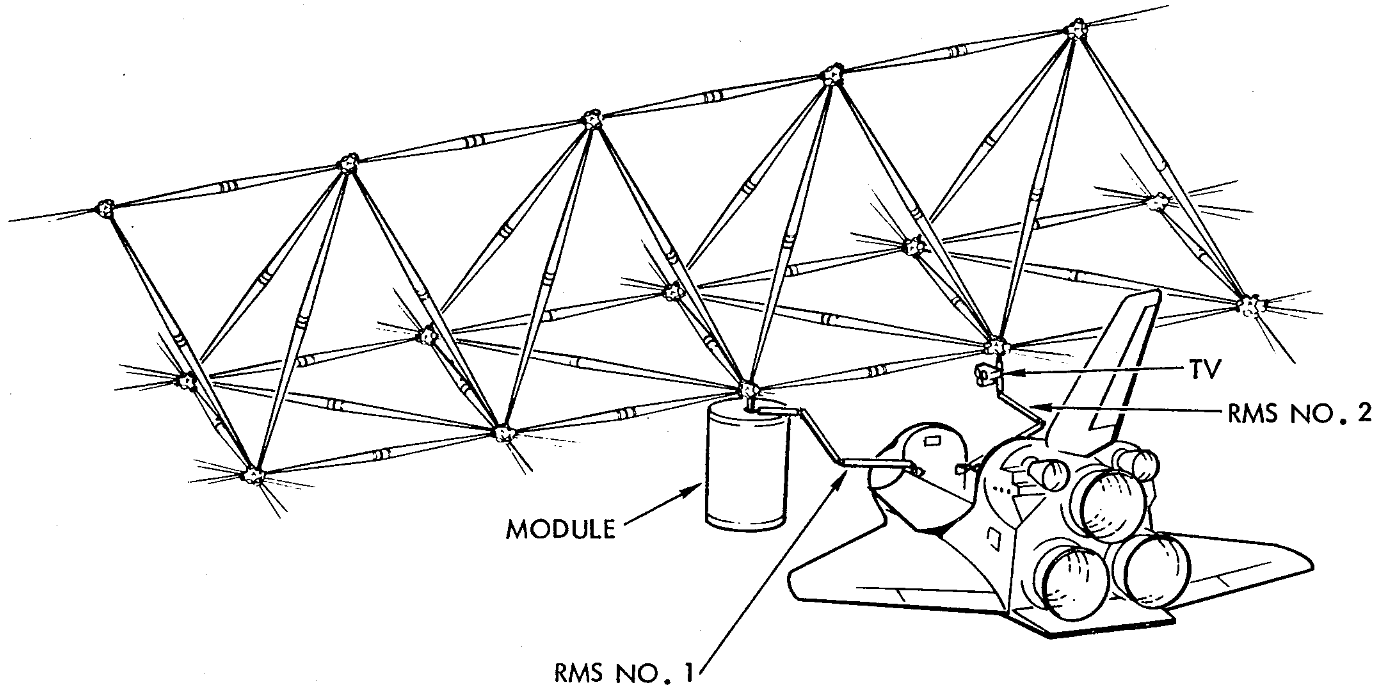


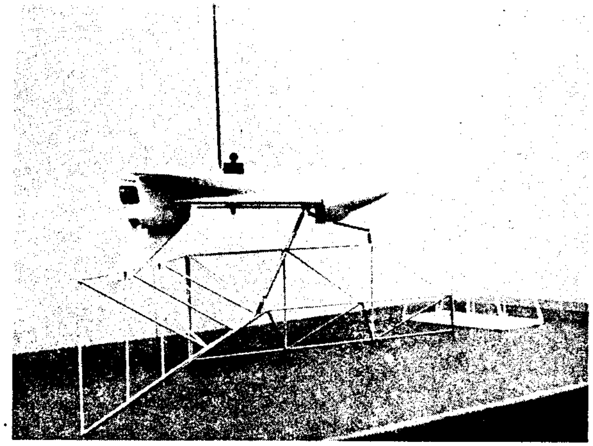
Chart 9

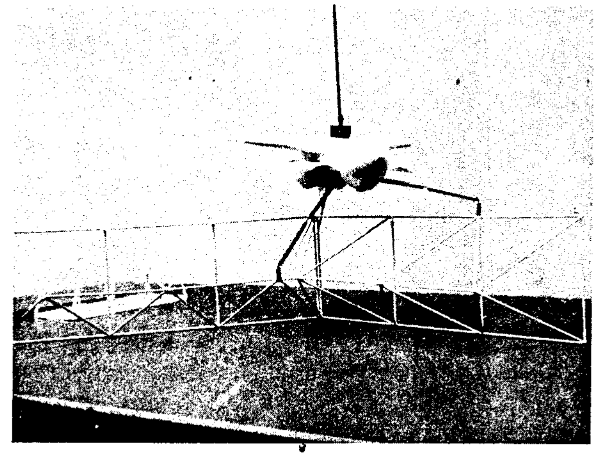
CHART 10 - WALKING TECHNIQUE

This chart illustrates the concept of "walking" the Orbiter--using two RMS arms. The photo insets show two views of a step in the action. For the case illustrated, the structure was a hexagonal frame made up of 14-m (45-ft) length struts; the concept would, however, apply to other structures insofar as the structural nodes would be within the reach of the two arms. In the position shown, the Orbiter has just engaged a forward node with its right-hand arm. The subsequent action would be to disengage the left-hand arm and, using the engaged RMS, maneuver the Orbiter and its free (left-hand) arm to reach the next node. These operations would be, of course, under control of the crew, and the Orbiter's orientation would be such as to assure safe clearance at all times.

The lower graph shows the approximate rate at which the Orbiter could translate with respect to the structure. To achieve this rate, the loads placed on the unions/nodes would be well under 100 N-m/ft-lb. Even at these low rates, the total time required for all walking operations for a 100-m structure could be less than a small fraction of one day's time out of the total mission.

WALKING TECHNIQUE





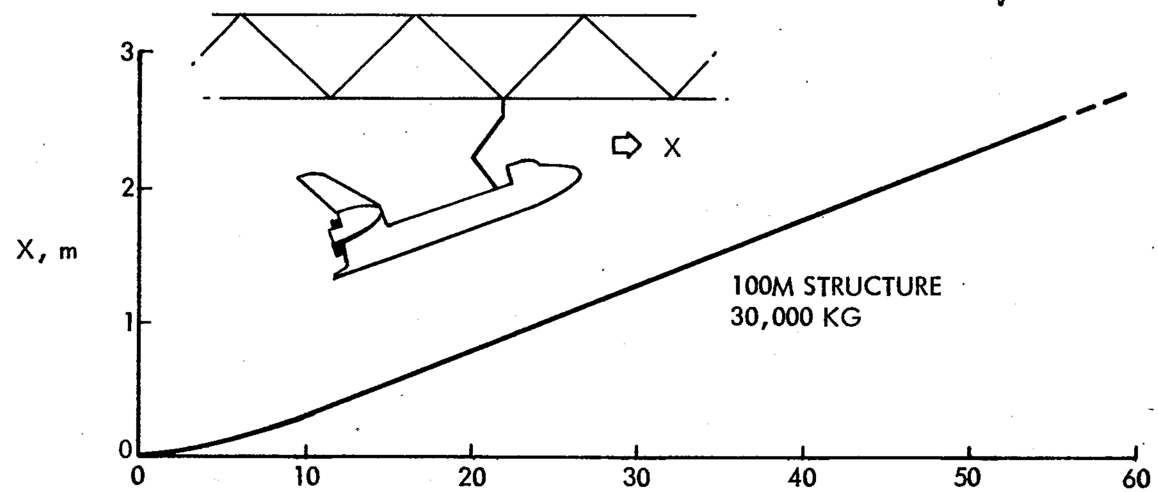


Chart 10

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Space Division

CHART 11 - RMS ISSUES

The previous charts have shown construction operations using the Orbiter's RMS in various ways--but always within its currently specified limits. This chart identifies some of the key issues underlying the feasibility of using the RMS as postulated. The reach of the 15-m (50-ft) long RMS is adequate for most foreseen (early) operations, but additional kinematic investigations are required. The accuracy with which the manually controlled RMS can position an element is still unknown, but on-going simulations at JSC and SPAR are encouraging. The principal question is, perhaps, how much <u>time</u> it will take to make an engagement--not <u>whether</u> an engagement can be made. In some instances, control of two arms may be desirable; more study is required to fully assess the impact upon software and control station provisions. The dynamics of the total Orbiter-platform-RMS system must be evaluated and simulated in detail to guard against undesirable oscillations and dynamic loads. Visibility requirements in terms of field of view, lighting, depth of perception, and other factors must be determined. In addition, the peculiar demands of construction operations will require special end-effector designs and associated tests. The following two charts illustrate planned IR&D experiments which should shed additional light upon some of these questions.

RMS ISSUES

- ☆ REACH
- ☆ ACCURACY - TIME
- ☆ CONTROL
- ☆ DYNAMICS
- ☆ VISIBILITY
- ☆ EFFECTOR

Chart 11

Rockwell International
Space Division

CHART 12 - JOINING EXPERIMENT

This chart shows the concept of an experiment planned for early summer testing in JSC's Manipulator Development Facility (MDF). The test experiment would utilize the existing simulator/arm to evaluate accuracy-time relationships in making a structural joint. The model joint includes a ball-ended strut and a union with latching sockets to accept and retain the strut in place. The test variables would include strut orientation, approach aspect, and initiation distance. Although the test results would be obtained in a 1-g field, electronic simulations at JSC are expected to provide additional understanding of the gravity-dynamics factors.

JOINING EXPERIMENT (MDF)

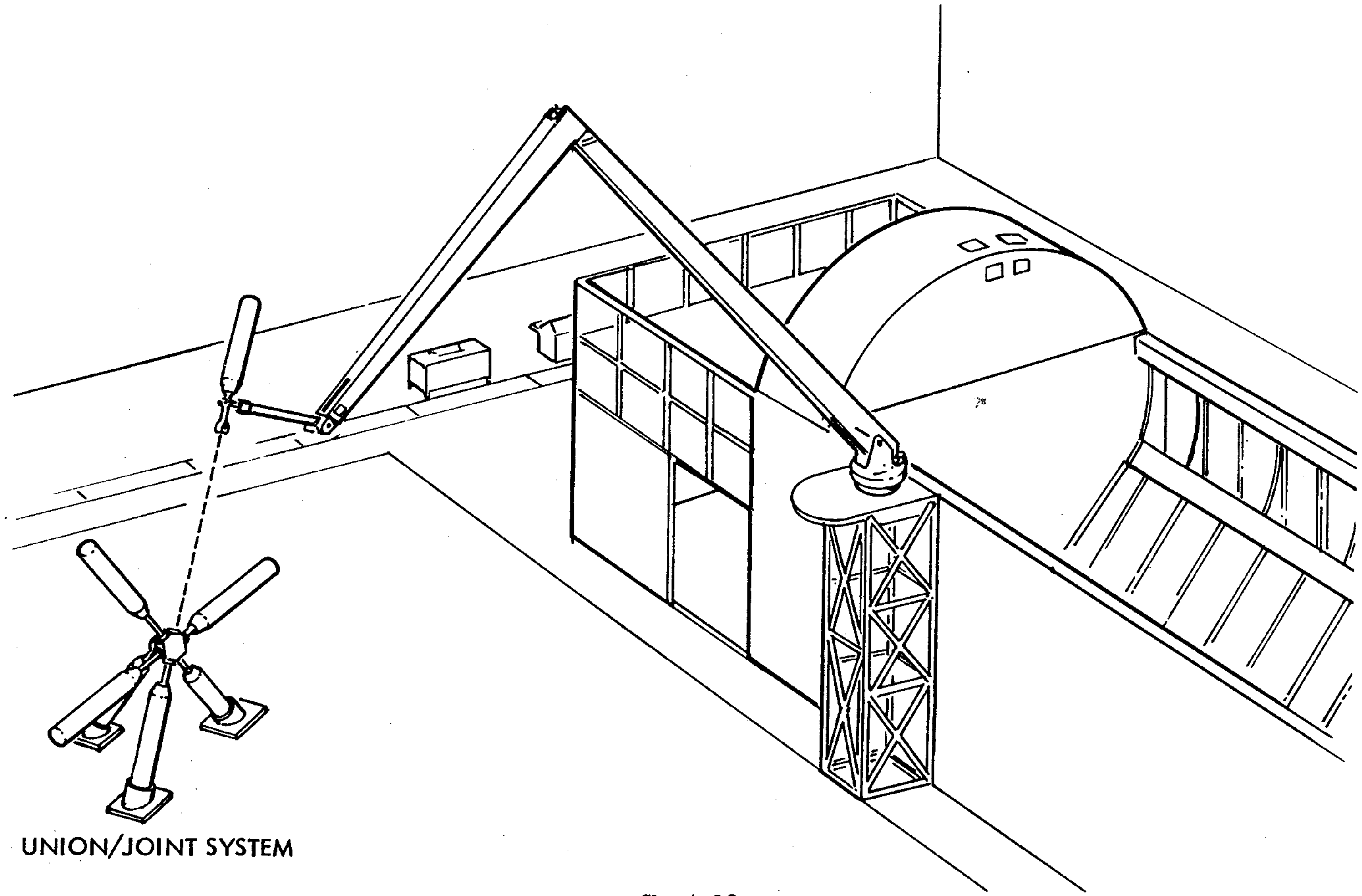

Chart 12

CHART 13 - CONNECTOR EXPERIMENT

A concept for an experiment in MSFC's Neutral Buoyancy Facility (NBF) is shown here. In this experiment, also planned for early summer testing, the capability of an EVA astronaut to make and unmake an electrical connector, similar to that previously described, would be demonstrated. The test model would include both halves of the connector and a harness to provide simulation of handling and bending restraints. The connector would also include a small internal battery circuit to provide visual proof of continuity across all pins.

Upon completion of the NBF/EVA tests, it is planned that a test setup, similar to that of the joining experiment, would be run at the MDF using the simulator arm to demonstrate the RMS mode of making the electrical connection. Although initial discussion on these tests have been held with JSC and MSFC, these experiments have not yet been approved for the MDF and NBF facilities.

CONNECTOR EXPERIMENT (NBF)

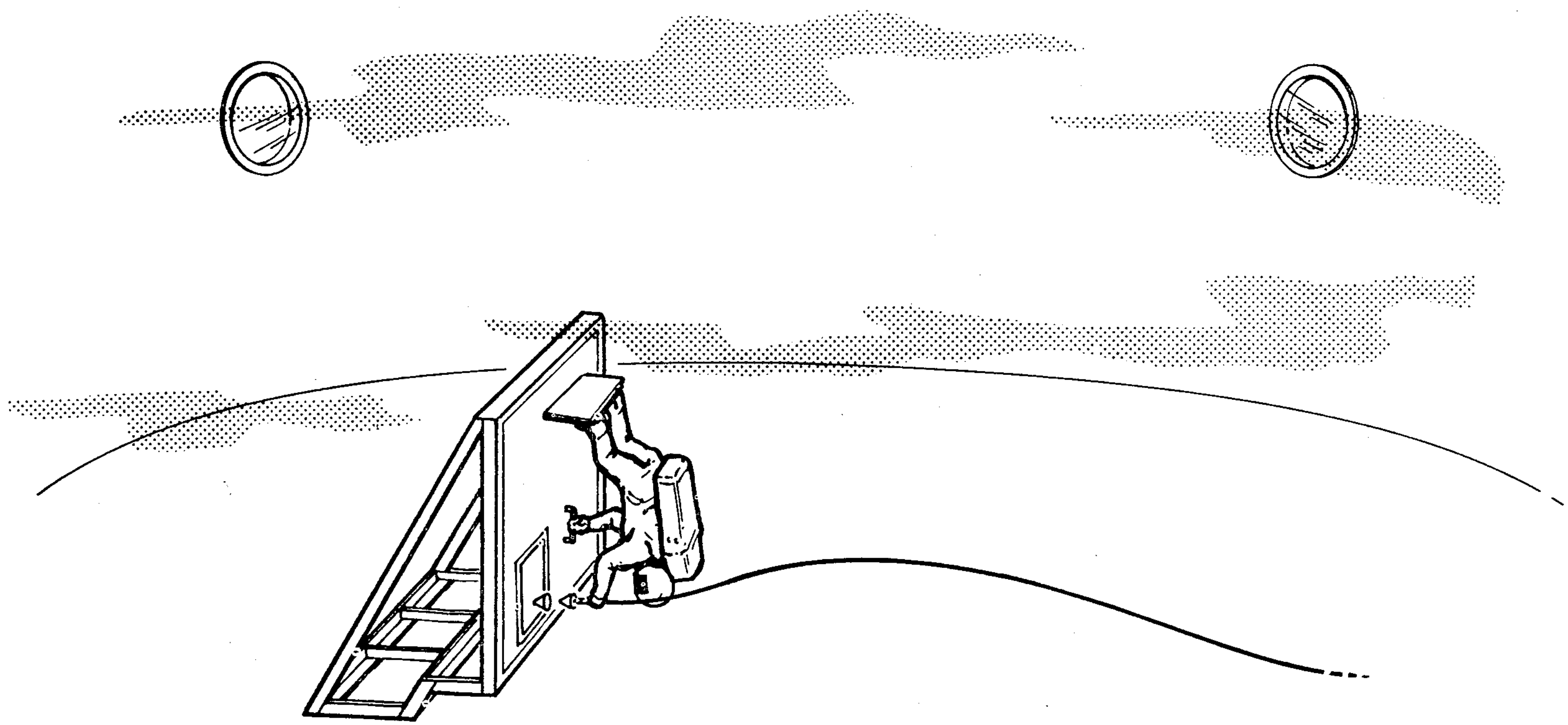

Chart 13

CHART 14 - CONCLUSIONS

The accompanying chart identifies the major points of this paper.

CONCLUSION

☆ PRELIMINARY CONCEPTS HAVE BEEN IDENTIFIED FOR EQUIPMENT INSTALLATION

- MECHANICAL
- ELECTRICAL
- ORBITER OPERATIONS

☆ STUDIES AND SIMULATIONS ARE REQUIRED FOR PROOF-OF-CONCEPT

☆ FY 1978 IR&D EXPERIMENTS ARE PLANNED TO OBTAIN INITIAL INSIGHTS

Chart 14

Rockwell International
Space Division

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Equipment Installation on Large Space Structures

1. Dynamic Considerations for the Orbiter-Attached-to-Structure During Erection Operations

The present analyses see no reasons why the erection operations could not proceed with the reaction control system in the Orbiter turned off. A partial structure (no RCS) would move in response to erection operations, but the predicted rates will allow maintaining the relative geometry between the Orbiter and the structure during operations where the Orbiter "walks" along the structure. The force capability of the Manipulator System appears capable of maintaining the "tail down" attitude of the Orbiter.

2. Plans for a Dual Manipulator System

OFT-3 will be the first flight with any manipulator system. The addition of a second manipulator awaits a subsequent program decision on the part of the NASA and the OMB.

Structural/Thermal Considerations For Design Of Large Space Platform Structures

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Presented At
Large Space Systems Technology Seminar
Langley Research Center

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STRUCTURAL/THERMAL CONSIDERATIONS FOR DESIGN OF LARGE SPACE PLATFORM STRUCTURES

BY

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INTRODUCTION

PLACING A LARGE, STS-COMPATIBLE PLATFORM ON ORBIT CAN BE ACCOMPLISHED WITH A CONSTRUCTION METHOD EMPLOYING BOTH DEPLOYABLE AND ERECTABLE STRUCTURES. THIS PAPER DISCUSSES SUCH A PLATFORM, A MULTIFUNCTIONAL MECHANISM FOR DEPLOYABLE STRUCTURES, AND AN ON-ORBIT ASSEMBLY TECHNIQUE FOR ERECTABLE STRUCTURES.

CONFIGURATION CONTROL OF LARGE ON-ORBIT STRUCTURES IS DEPENDENT IN PART ON THE RESPONSE OF THE STRUCTURE TO THE NATURAL THERMAL RADIATION AND TO ON-BOARD HEATING. THIS PAPER DISCUSSES ANALYSES WHICH ASSESS THE THERMAL DISTORTION OF A SIMPLE OPEN TRUSS AND A MORE COMPLEX TRUSS.

STRUCTURAL/THERMAL CONSIDERATIONS (Figure 1)

DEPLOYABLE AND ERECTABLE LARGE TRUSS STRUCTURES WERE INVESTIGATED. A SCENARIO WAS DEVELOPED WHICH WOULD ENABLE THE PLACING OF A 300 M x 300 M PLATFORM IN LOW EARTH ORBIT USING THE STS. THE PLATFORM IS ASSEMBLED USING 7 DEPLOYABLE TETRAHEDRAL MODULES, 120 M x 104 M IN SIZE, MOUNTED (3 POINT SUSPENSION) TO AN ERECTABLE TRUSS STRUCTURE CONSISTING OF 102 COLUMNS.

A KNEE JOINT CONCEPT, APPLICABLE TO TYPICAL DEPLOYABLE TRUSS STRUCTURES AND CONTAINING A DEPLOYMENT, LATCHING, AND DAMPING MECHANISM SYSTEM, WAS SELECTED FOR DETAIL INVESTIGATION.

AN ON-ORBIT ASSEMBLY PROCEDURE WAS IDENTIFIED FOR ERECTABLE STRUCTURES ALONG WITH COMPATIBLE ASSEMBLY JOINTS.

THE STRUCTURAL RESPONSE WAS DETERMINED FOR (1) A SIMPLE OPEN TRUSS SUBJECTED TO THE NATURAL THERMAL RADIATION AND (2) A MORE COMPLEX TRUSS SUBJECTED TO ON-BOARD HEATING AS WELL AS THE NATURAL THERMAL RADIATION.

Structural/Thermal Considerations

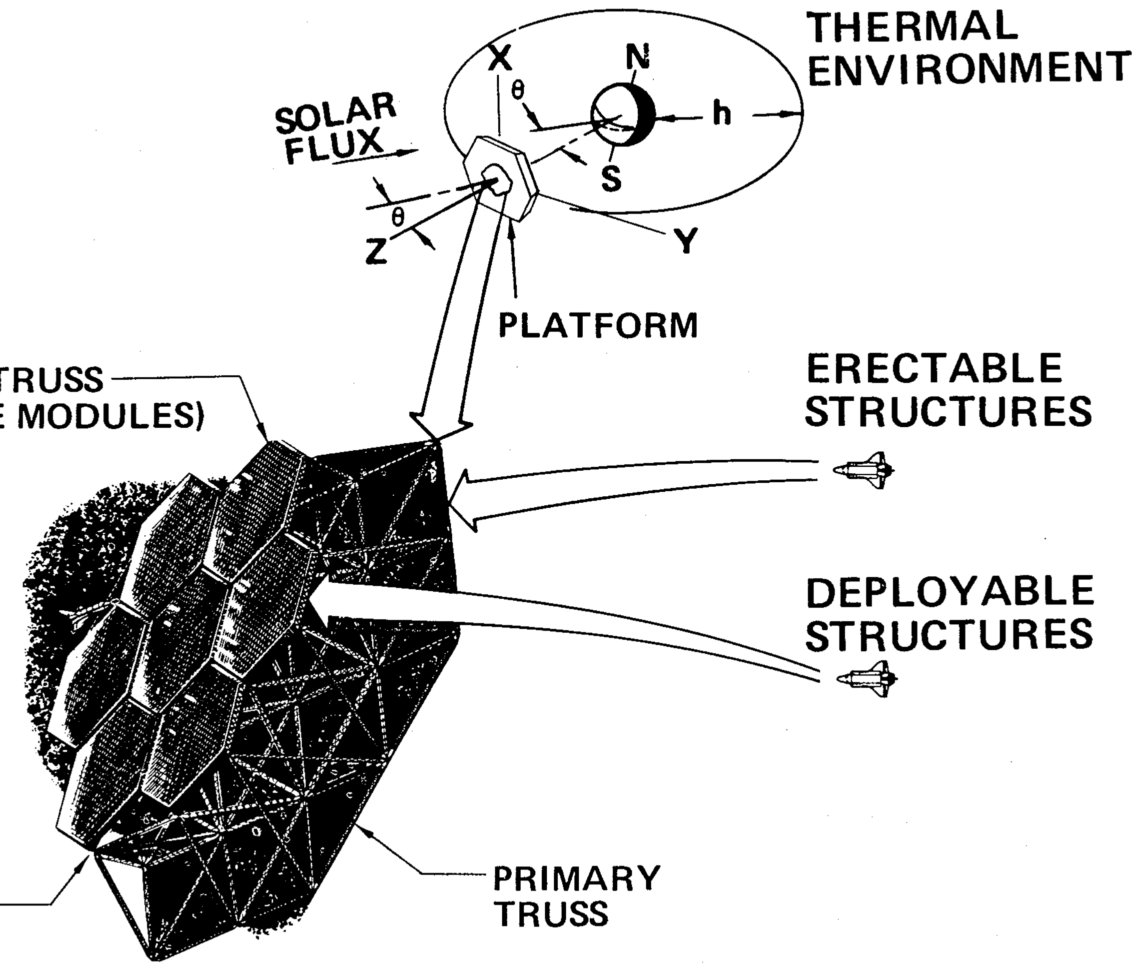


Figure 1

DEPLOYABLE STRUCTURE JOINTS (Figure 2)

THE USE OF STORED INTERNAL ENERGY FOR TRUSS DEPLOYMENT WAS SELECTED OVER OTHER OPTIONS, E.G., SPIN ABOUT Z AXIS, INFLATION/PNEUMATIC TECHNIQUES, EXPANSION BY RADIAL THRUSTER, TENSION CABLES WITH PULLEYS. MODELS WERE FABRICATED TO DEMONSTRATE DEPLOYMENT/PACKAGING TECHNIQUES. A CLUSTER FITTING JOINT DESIGN WAS SELECTED. AN EVALUATION OF KNEE JOINT CONCEPTS WHICH CONTAIN STORED INTERNAL ENERGY WAS MADE AND A CANDIDATE SELECTED FOR DETAIL INVESTIGATION.

Deployable Structure Joints

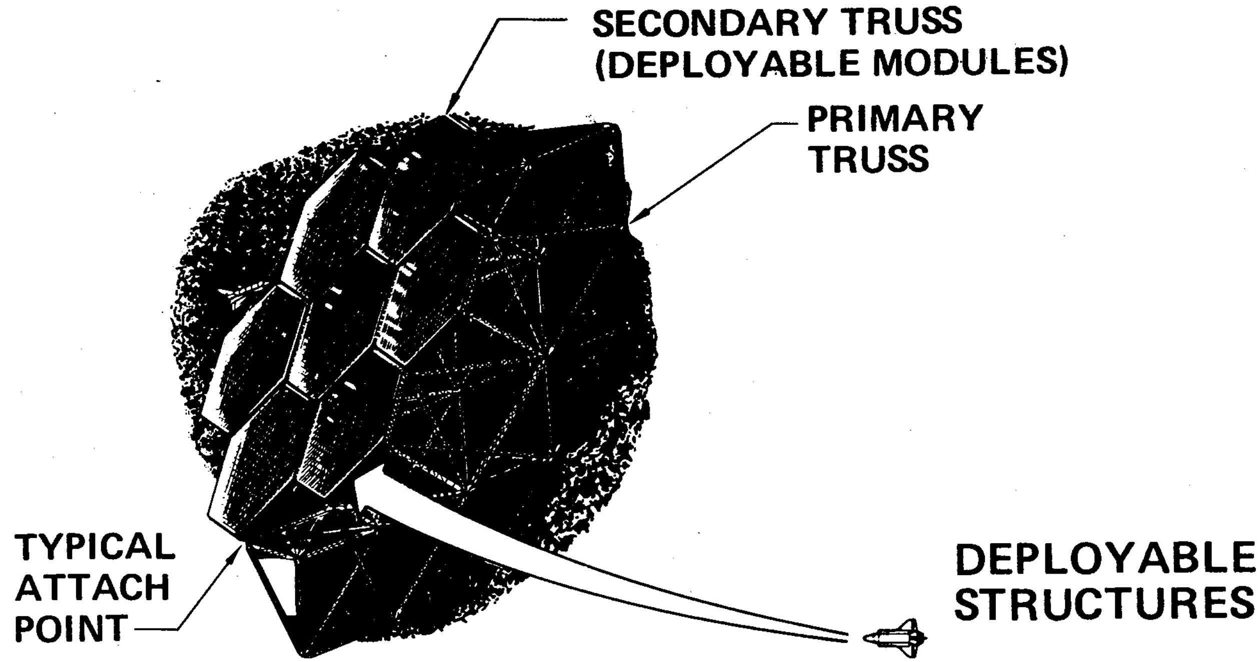
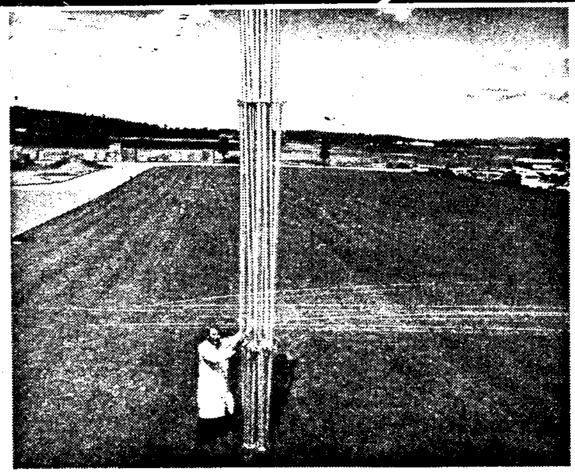


Figure 2

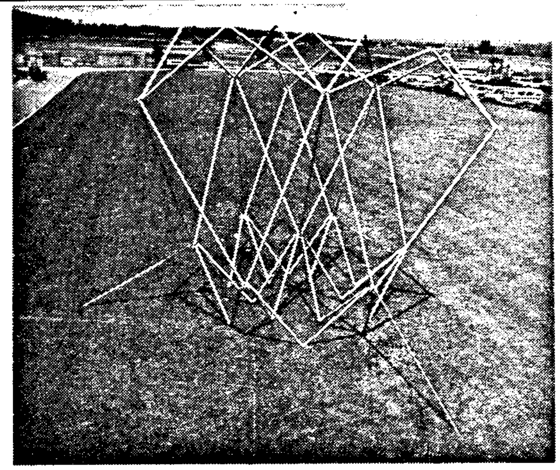
TYPICAL DEPLOYABLE TRUSS MODULE (Figure 3)

A 36-ELEMENT TETRAHEDRAL TRUSS MODULE WAS FABRICATED TO SERVE AS (1) A MODEL FOR DEMONSTRATING THE PACKAGING AND DEPLOYMENT CONCEPT AND (2) A MOCKUP TO EVALUATE JOINT CONCEPTS. THE 50 MM DIAMETER ALUMINUM ALLOY STRUT MEMBERS ARE 3 M LONG. PACKAGED LENGTH OF THE TRUSS IS 6 M. WHEN DEPLOYED, THE MAIN SURFACE DIMENSION OF THE TRUSS IS 6 M AND THE DEPTH IS 2.5 M.

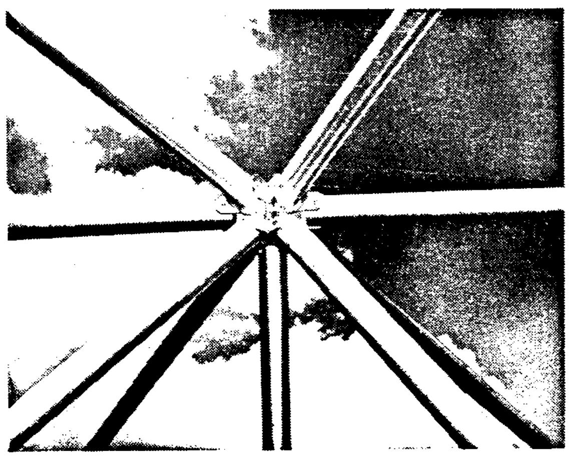
Typical Deployable Truss Module



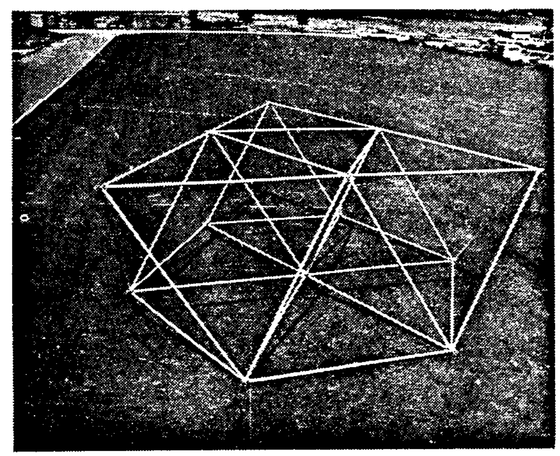
a) PACKAGED



b) PARTIALLY DEPLOYED



c) CLUSTER FITTING



d) DEPLOYED

Figure 3

CLUSTER FITTING JOINT (Figure 4)

CONCEPTS FOR THE CLUSTER FITTINGS WHICH JOIN THE SURFACE AND THE INTERSURFACE MEMBERS WERE EVALUATED. FITTINGS WITH SINGLE LUGS FOR THE PIN JOINT CONNECTIONS WERE FOUND TO HAVE CONSIDERABLE JOINT FREE PLAY WHEN THE ATTACHING MEMBERS WERE IN THE PACKAGED CONDITION OR ARTICULATING FOR DEPLOYMENT. A QUALITATIVE EVALUATION SUGGESTED THAT THIS FREE PLAY WOULD DETRACT SUBSTANTIALLY FROM THE CAPABILITY OF THE TRUSS TO DEPLOY UNIFORMLY, IN A SMOOTH FASHION, AND WITHOUT EXCESSIVE BINDING AT THE JOINTS. NON-UNIFORM DEPLOYMENT AND BINDING AT THE PIN JOINT CONNECTIONS COULD RESULT IN HIGH MEMBER LOADS AND REQUIRE AN INCREASE IN STORED DEPLOYMENT ENERGY. FITTINGS WITH DOUBLE LUGS FOR THE PIN JOINT CONNECTIONS HAVE MUCH LESS JOINT FREE PLAY THAN THE SINGLE LUG FITTINGS AND THEREFORE WERE SELECTED.

Cluster Fitting Joint

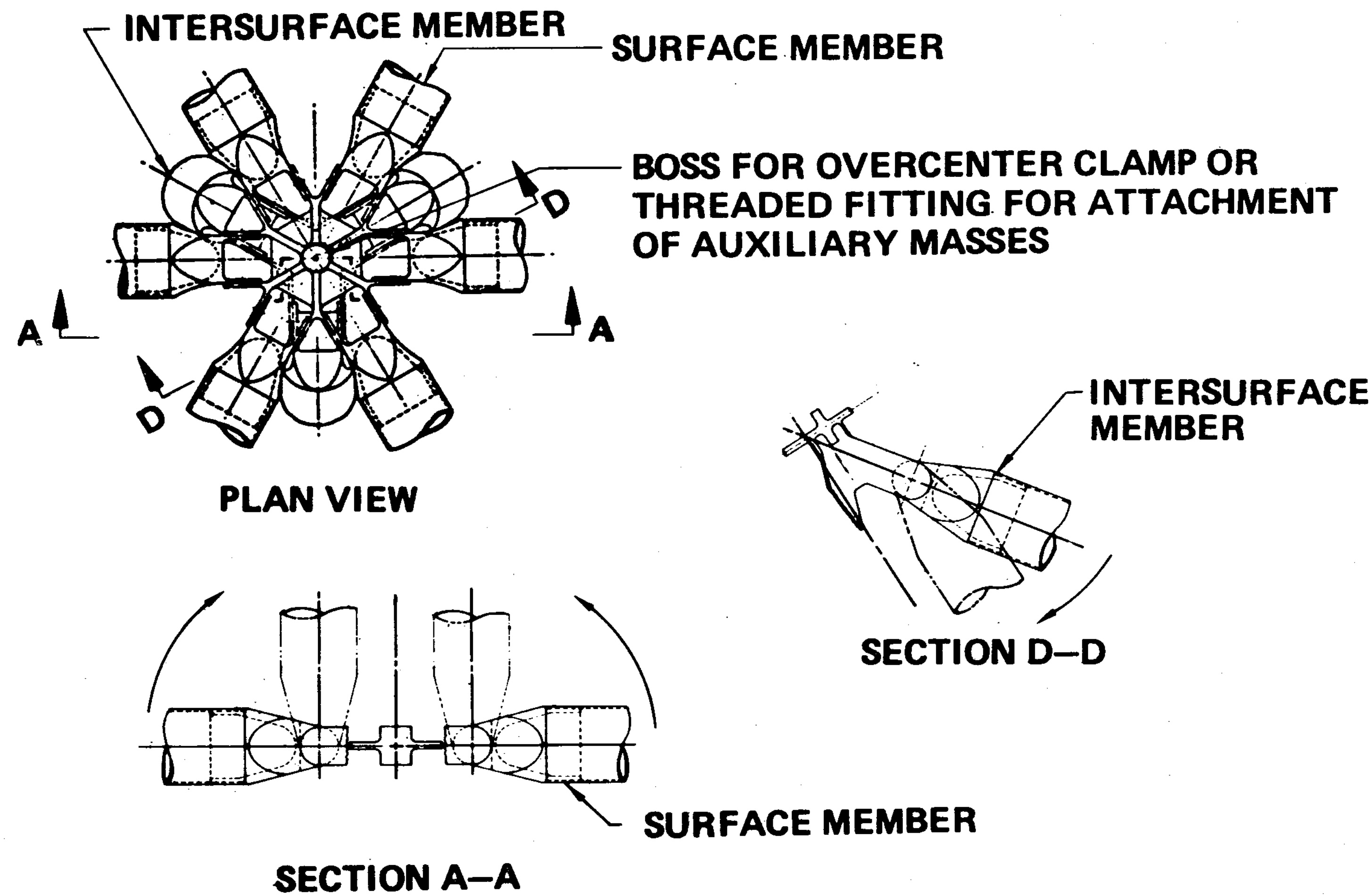


Figure 4

KNEE JOINT CONCEPT EVALUATION (Figure 5)

THREE KNEE JOINT CONCEPTS WHICH STORE INTERNAL ENERGY FOR TRUSS DEPLOYMENT WERE EVALUATED. THE EVALUATION CRITERIA INCLUDED (1) THE CAPABILITY TO ADJUST THE AMOUNT OF STORED ENERGY SO THAT THE ENERGY VALUES AT EACH TRUSS JOINT CAN BE TAILORED AS REQUIRED FOR A RELIABLE DEPLOYMENT CYCLE WITH MINIMUM LOADS ON THE MEMBERS, (2) ZERO FREE PLAY IN THE JOINT OR A MECHANISM TO REDUCE FREE PLAY, (3) A MECHANISM TO ABSORB THE SYSTEM'S ENERGY DURING THE FINAL STAGES OF DEPLOYMENT, AND (4) A CONVENIENT METHOD FOR RELEASING THE LOCKING MECHANISM OF THE DEPLOYED JOINT TO PERMIT REPACKAGING OF THE TRUSS. THE CAPABILITY TO CONVENIENTLY REPACKAGE A DEPLOYED TRUSS IS NECESSARY DURING THE GROUND TEST PHASE OF A DEVELOPMENT TEST PROGRAM AT WHICH TIME NUMEROUS DEPLOYMENT TESTS WILL BE CONDUCTED ON THE TRUSS. IN THE EVALUATION, THE TORQUE SPRING/CAM LATCH CONCEPT WAS RATED THE HIGHEST.

Knee Joint Concept Evaluation

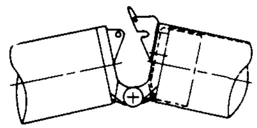
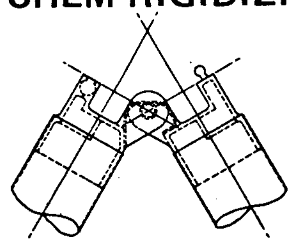
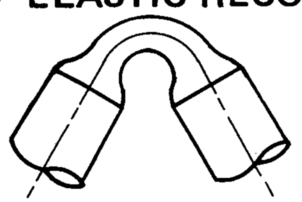
CONCEPT	EVALUATION CRITERIA				
	DEPLOYMENT ENERGY	ADJUSTABLE DEPLOYMENT ENERGY	FREE-PLAY REDUCING MECHANISM	DAMPING MECHANISM	CONVENIENT RELEASE METHOD
a) TORQUE SPRING/ CAM LATCH 	√	√	√	√	√
b) TORQUE SPRING/ CHEM RIGIDIZING 	√	√	√	√	
c) ELASTIC RECOVERY 	√		√		

Figure 5

FEATURES OF TORQUE SPRING/CAM LATCH CONCEPT (Figure 6)

THIS CONCEPT CONTAINS TWO DEPLOYMENT SPRINGS. THESE SPRINGS COUPLED WITH ALL THE OTHER KNEE JOINT SPRINGS PROVIDE THE ENERGY FOR TRUSS DEPLOYMENT. BUT AS WELL, THESE SPRINGS PROVIDE LOCAL ENERGY TO DRIVE THE LATCH PIN AGAINST THE CAM, PUSHING IT INTO THE "LATCHED" POSITION. THE USE OF TWO SPRINGS AT EACH JOINT MINIMIZES TORSION IN THE TUBULAR MEMBERS AND PROVIDES DEPLOYMENT/LATCHING RELIABILITY IN CASE ONE SPRING FAILS. THE CAM SURFACE IS SHAPED AND ARRANGED WITH ITS FULCRUM AND COMPRESSION SPRING SUCH THAT IN THE "LATCHED" POSITION, IF THERE IS FREE-PLAY OF THE LATCH PIN IN THE SLOT, ANY JOINT VIBRATION WILL DRIVE THE CAM AGAINST THE PIN, FORCING THE PIN AGAINST THE SLOT.

TO REPACKAGE THE TRUSS, THE CAM LEVER IS PUSHED IN A DIRECTION AWAY FROM THE MEMBER ON WHICH THE CAM BRACKET IS MOUNTED. THIS ACTION MOVES THE CAM SURFACE PAST THE LATCH PIN (FURTHER COMPRESSING THE LATCH SPRING). A HOLDING PIN IS THEN INSERTED WHICH KEEPS THE CAM IN THE "RELEASED LATCH" POSITION. THE NEXT STEP IS TO RELEASE THE TORSION IN THE SPRINGS BY REMOVING THE SPRING LEG RETAINING SCREW. THIS CAUSES THE SPRING TO UNWIND 360°. THEN THERE IS A MINIMUM RESISTANCE AGAINST ARTICULATING THE SURFACE MEMBER INTO THE PACKAGED CONDITION.

Features of Torque Spring/ Cam Latch Concept

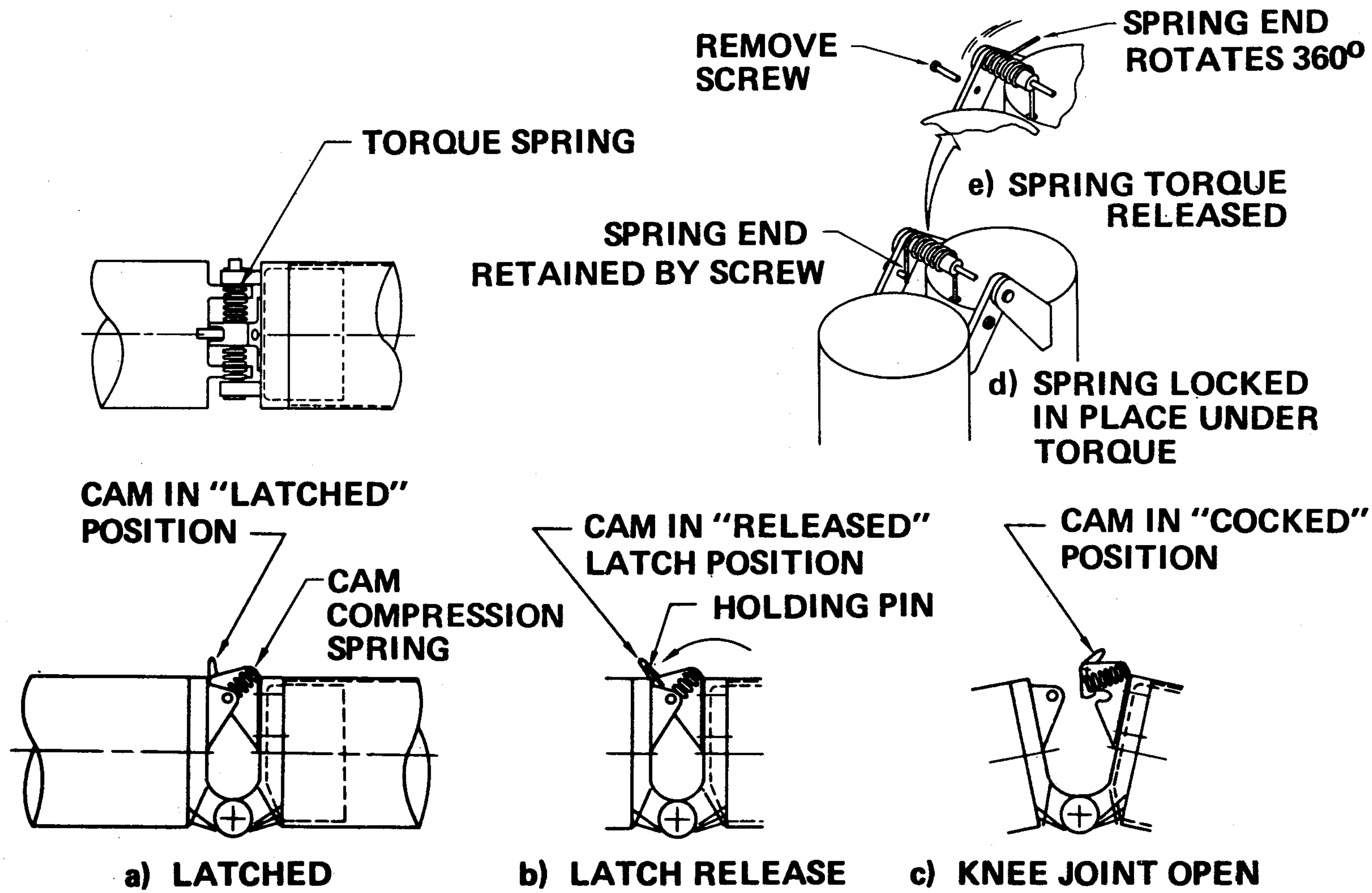


Figure 6

JOINT AND ASSEMBLY CONCEPTS FOR ERECTABLE SPACE STRUCTURES (Figure 7)

THE ASSEMBLY SCENARIO FOR THE TWO-TIER PLATFORM ILLUSTRATED IN THE "ERECTABLE STRUCTURE" CHART IS INITIATED WITH THE SHIPPING TO ORBIT, DEPLOYMENT, AND DOCKING OF THE SECONDARY TRUSS UNITS. THE SECONDARY TRUSS IS THEN USED AS A COLUMN ASSEMBLY PLATFORM. STRUT ASSEMBLY MACHINES ARE MOUNTED TO THE PLATFORM, RECEIVE STRUT MAGAZINES, AND ASSEMBLE THE 10-METER-LONG STRUTS. BY MEANS OF MANIPULATORS AND EVA ACTIVITY THE 130-METER-LONG PRIMARY TRUSS COLUMNS ARE FABRICATED FROM THE STRUTS. THE COMPLETED COLUMNS ARE ASSEMBLED TOGETHER TO FORM THE PRIMARY TRUSS AND CONNECTED AT THE APPROPRIATE POINTS TO THE SECONDARY TRUSS UNITS. THE FINAL TWO-TIER PLATFORM ASSEMBLY OPERATION IS TO DISENGAGE THE DOCKING DEVICES CONNECTING THE SECONDARY TRUSS UNITS. THIS ALLOWS FOR LEVELLING ADJUSTMENT OF THE SECONDARY TRUSS STRUCTURE MODULES.

Joint and Assembly Concepts for Erectable Space Structures

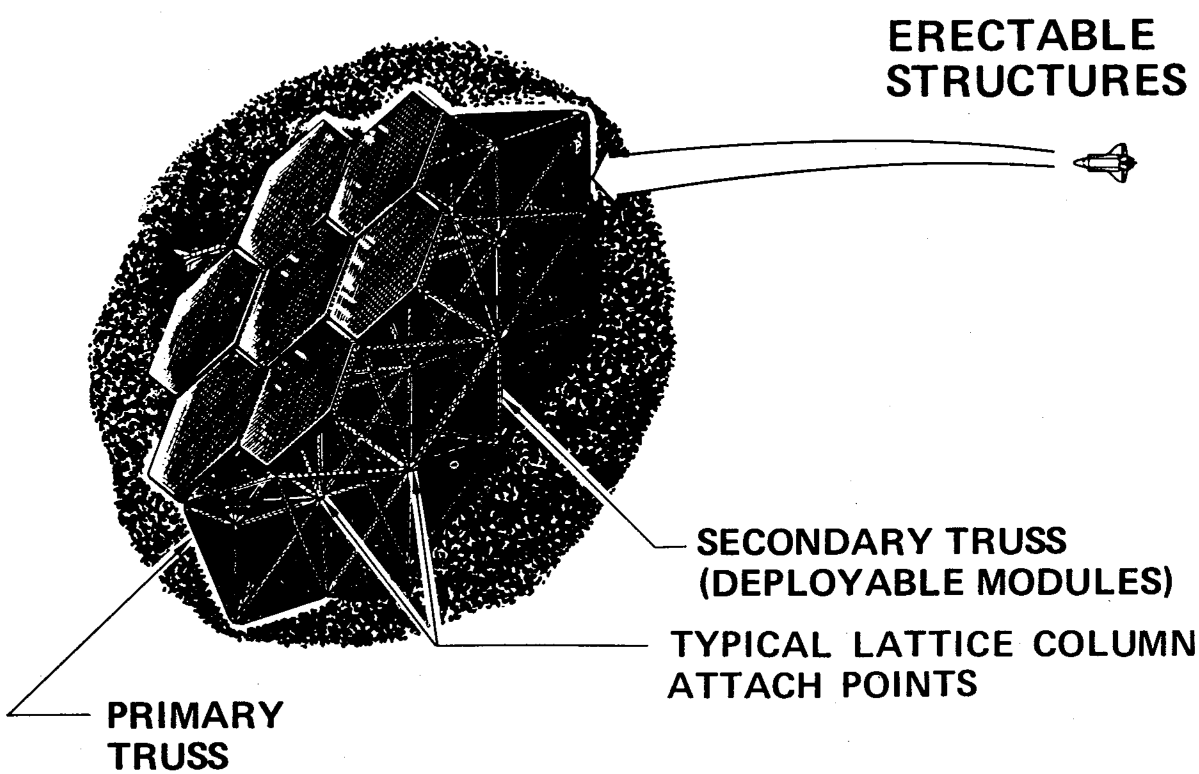


Figure 7

PRIMARY STRUCTURE - ERECTABLE LATTICE COLUMNS (Figure 8)

THE PRIMARY STRUCTURE SHOWN IN THE "ERECTABLE STRUCTURE" CHART CONSISTS OF 102 LATTICE COLUMNS, 130 METERS LONG. TO BE SHUTTLE COMPATIBLE, THESE COLUMNS ARE CONSTRUCTED FROM 111 REPEATING TAPERED COLUMN ELEMENTS, EACH 10 METERS LONG.

A BALL JOINT CONNECTION WHICH ALLOWS MULTIPLE COLUMNS TO BE FITTED AT THE NODAL POINTS WAS ASSUMED AS THE MAIN JOINING METHOD.

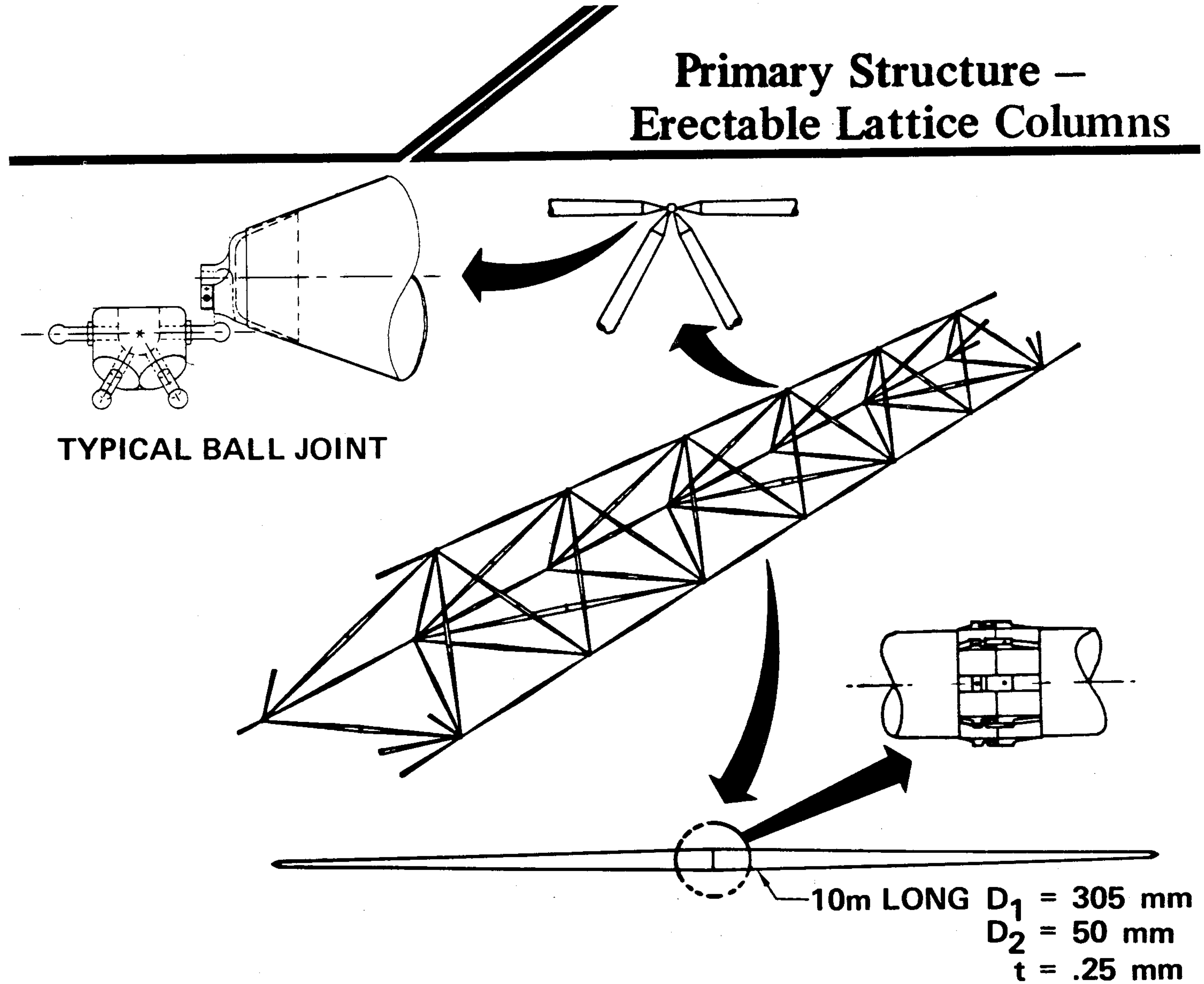

Primary Structure –
Erectable Lattice Columns
TYPICAL BALL JOINT
10m LONG D_1 = 305 mm
D_2 = 50 mm
t = .25 mm

Figure 8

THERMAL DISTORTION ANALYSES OF LARGE TRUSS STRUCTURES (Figure 9)

TEMPERATURE DISTRIBUTIONS FOR A SIMPLE OPEN TRUSS SUBJECTED TO NATURAL THERMAL RADIATION ONLY AND FOR A MORE COMPLEX TWO-TIER CONFIGURATION WITH REFLECTING SURFACES, AND SOLAR PLUS ON-BOARD HEATING WERE COMPUTED AND USED TO PREDICT GEOMETRIC DISTORTIONS FOR THE PLATFORM.

Thermal Distortion Analyses of Large Truss Structures

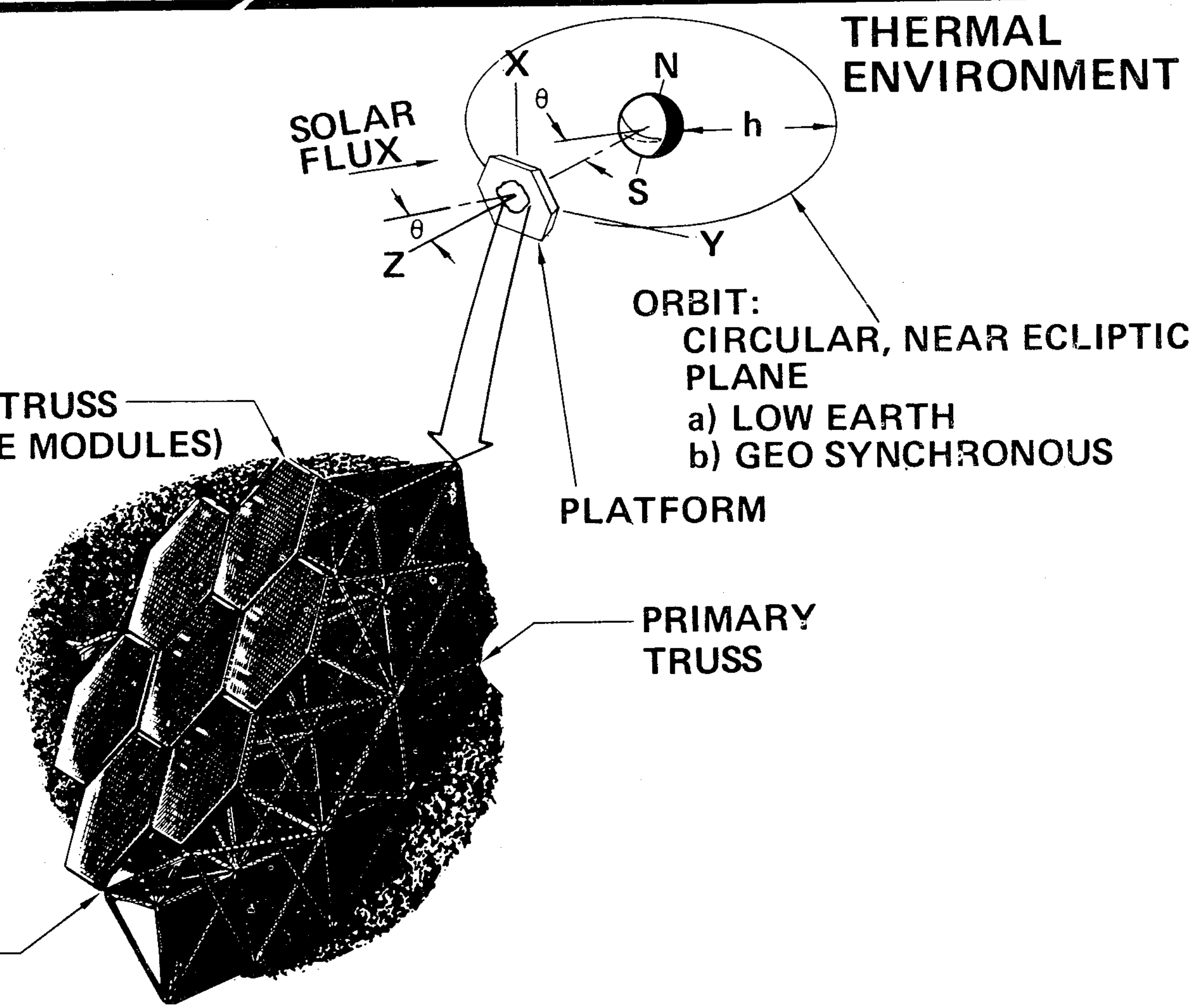
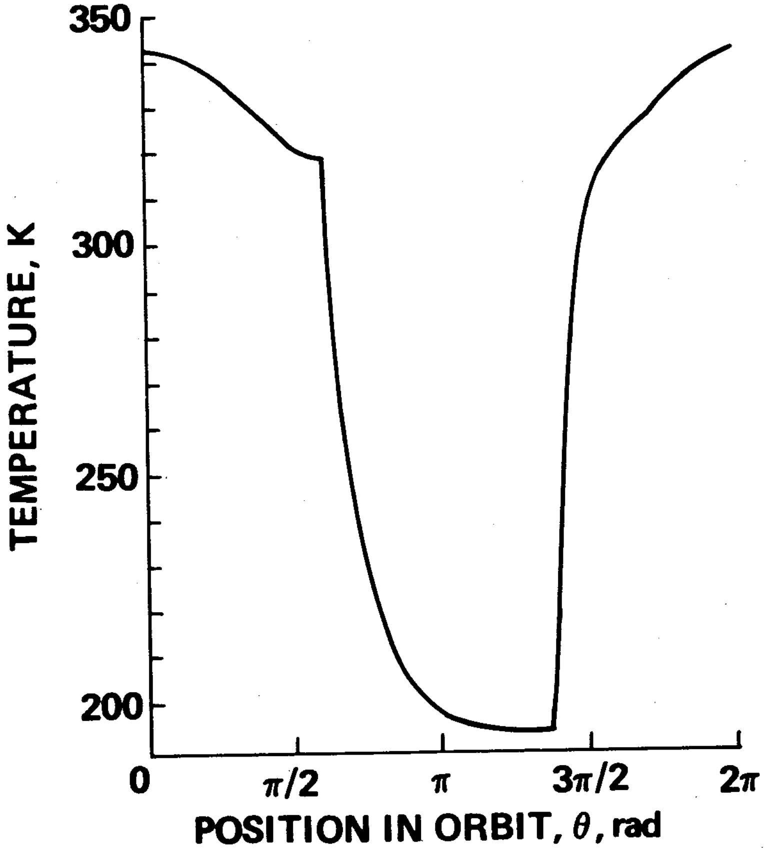


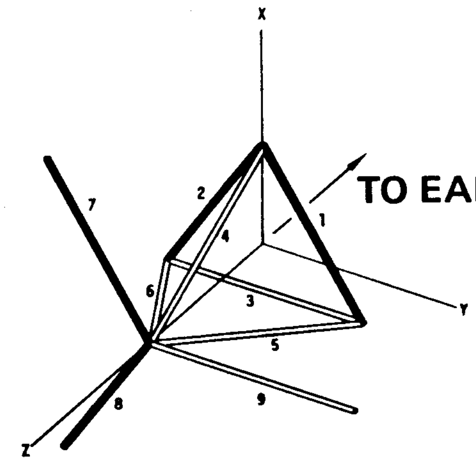
Figure 9

SAMPLE TEMPERATURES - TETRAHEDRAL MODULE (Figure 10)

A HEXAGONAL PLATFORM CONSTRUCTED FROM REPEATING TETRAHEDRAL MODULES WAS ANALYZED FOR TEMPERATURES AND THERMAL DISTORTIONS WHILE IN LOW EARTH ORBIT. THE THERMAL ENVIRONMENT CONSISTED OF SOLAR, EARTH-EMITTED, AND EARTH-REFLECTED RADIATION. TRANSIENT RESPONSE, I.E., THERMAL CAPACITANCE, OF THE MEMBERS WAS CONSIDERED BUT CONDUCTION THROUGH JOINTS AND RADIANT INTERCHANGE BETWEEN MEMBERS WERE IGNORED. A SEPARATE STUDY SHOWED JOINT CONDUCTANCE TO BE INSIGNIFICANT TO THE TEMPERATURES OF THE 50 MM DIAMETER GRAPHITE-EPOXY TUBES FOR ALL LENGTHS GREATER THAN ABOUT 3 METERS. CONFIGURATION SYMMETRY AND ORIENTATION IN ORBIT RESULTED IN CERTAIN MEMBERS, SUCH AS THOSE INDICATED IN THE SAMPLE PLOT, HAVING COMMON TEMPERATURE HISTORIES. THE PLOT SHOWS TEMPERATURES FOR ONE COMPLETE ORBIT, WITH THE EFFECT OF PASSAGE THROUGH THE EARTH'S SHADOW VERY EVIDENT.

Sample Temperatures - Tetrahedral Module





TETRAHEDRAL MODULE

TEMPERATURES SHOWN FOR MEMBERS 1, 2, 7, & 8

MEMBERS: GRAPHITE-EPOXY TUBES, d/t = 100

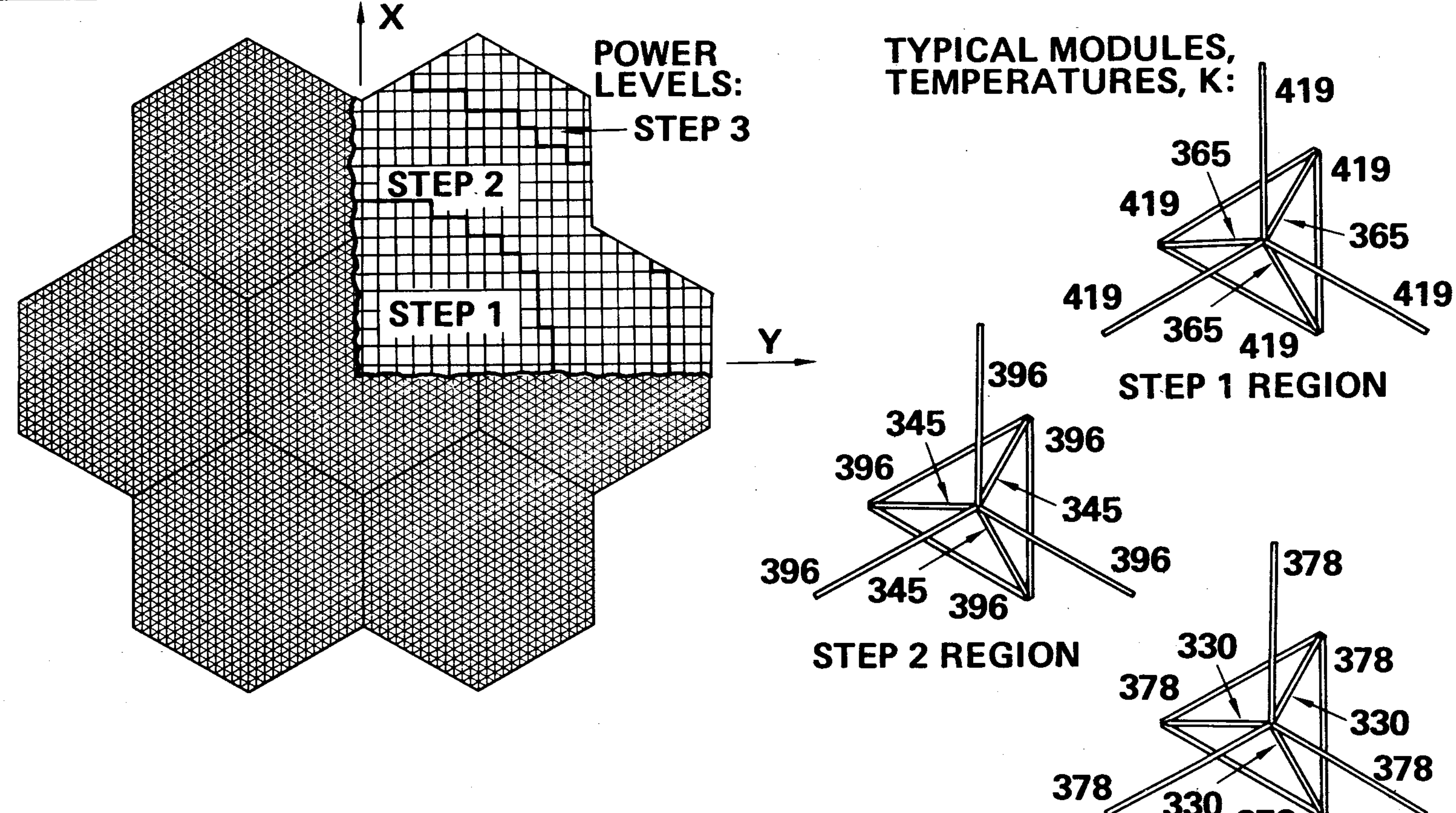
ORBIT: CIRCULAR, NEAR-ECLIPTIC PLANE, 463 km

Figure 10

TEMPERATURES OF SECONDARY TRUSS MEMBERS (Figure 11)

A TWO-TIER STRUCTURE, CONSISTING OF A HEXAGONAL PRIMARY TRUSS COMPOSED OF TWO RINGS OF TETRAHEDRAL MODULES, SUPPORTING 7 SECONDARY HEXAGONAL TRUSSES, WAS ANALYZED FOR THERMAL DISTORTIONS. EACH SECONDARY TRUSS CONSISTED OF SEVEN RINGS OF TETRAHEDRAL MODULES AND SUPPORTED ON ITS EARTH-FACING SIDE A CONTINUOUS ARRAY OF OPAQUE NON-STRUCTURAL PANELS. THESE PANELS WERE ASSUMED TO EMIT THERMAL RADIATION AT 3 DIFFERENT INTENSITIES (POWER LEVEL STEPS 1, 2, AND 3 ON THE DIAGRAM). THE STRUCTURE WAS ANALYZED IN A CIRCULAR, ECLIPTIC-PLANE GEOSYNCHRONOUS ORBIT AND TEMPERATURES WERE COMPUTED AT FIVE CONDITIONS: $\Theta = 0$, $\Theta = \Pi/2$ RAD, $\Theta = \Pi$ RAD WITH NO ECLIPSE, $\Theta = \Pi$ RAD WITH FULL ECLIPSE, AND $\Theta = 3\Pi/2$ RAD. THE CASE OF $\Theta = \Pi$ RAD WITH FULL ECLIPSE, WHEN THE RADIALLY-VARYING ON-BOARD HEATING TOTALLY DOMINATED THE TEMPERATURE DISTRIBUTIONS AND WAS FOUND TO BE THE MOST CRITICAL CASE FOR THERMAL DISTORTIONS. THE CHART SHOWS TEMPERATURES OF SECONDARY TRUSS MEMBERS FOR THIS CASE, FOR TETRAHEDRAL MODULES LOCATED WHOLLY WITHIN EACH OF THE 3 LEVELS OF ON-BOARD HEATING. THE PEAK ON-BOARD HEATING (STEP 1 REGION) WAS 3.5 KW/M^2 AT THE RADIATING PANELS. THE POWER STEP LEVELS USED WERE THE FIRST THREE STEPS OF A 10-STEP APPROXIMATION OF A 10 DB GAUSSIAN TAPER APPLIED TO THE STRUCTURE EXTENDED TO 1 KM DIAMETER.

Temperatures of Secondary Truss Members



ORBIT CONDITION:
$\theta = \pi$ rad , FULL ECLIPSE
ECLIPTIC PLANE GEOSYNCHRONOUS

Figure 11

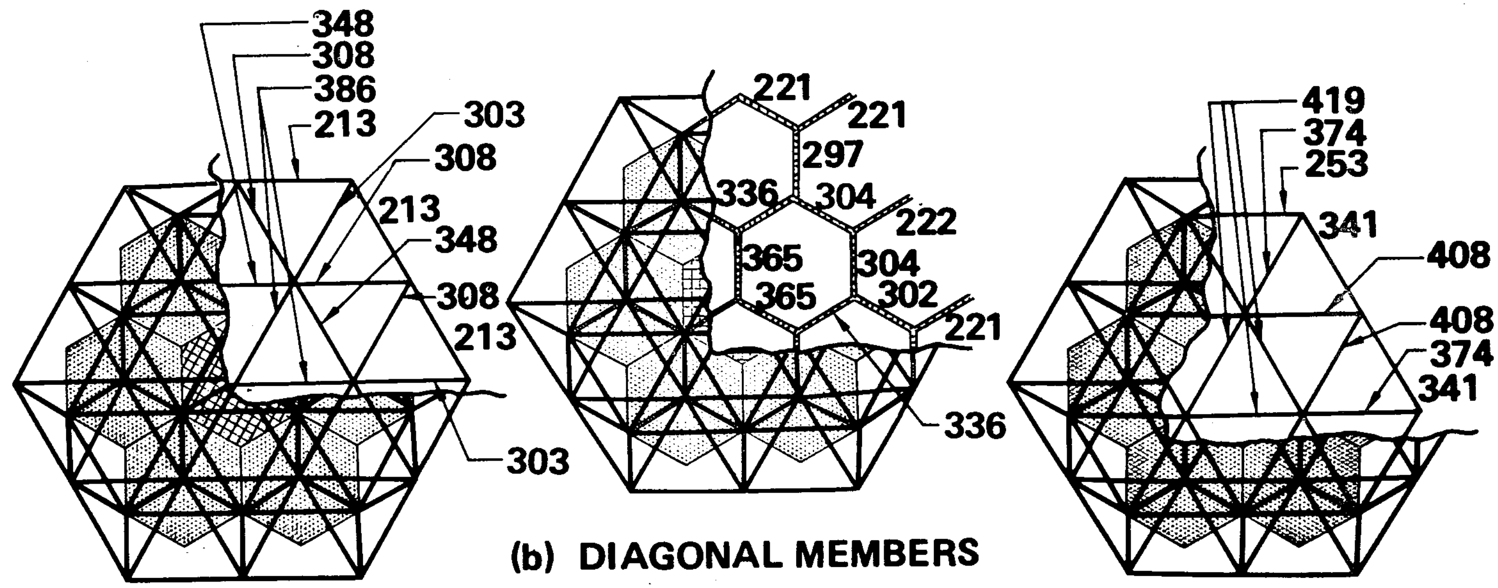
TEMPERATURES OF PRIMARY TRUSS MEMBERS (Figure 12)

THE CHART SHOWS SAMPLE TEMPERATURES FOR THE PRIMARY STRUCTURE OF THE TWO-TIER PLATFORM. AS FOR THE SECONDARY TRUSS MEMBERS, THE CASE OF $\Theta = \Pi$ RAD WITH FULL ECLIPSE RESULTED IN THE GREATEST TEMPERATURE DIFFERENTIALS AND THE MAXIMUM DISTORTIONS. FOR THIS CONDITION, THE ON-BOARD HEATING IS THE ONLY THERMAL INFLUENCE. THIS FACT, PLUS THE NEAR-CONCENTRIC BOUNDARIES OF POWER LEVEL STEPS, RESULTS IN TEMPERATURE DISTRIBUTIONS WITH AXIAL SYMMETRY ABOUT THE Z-AXIS. TEMPERATURES FOR THE TWO-TIER PLATFORM, LIKE THOSE FOR THE STRUCTURE OF "SAMPLE TEMPERATURES - TETRAHEDRAL MODULE" WERE COMPUTED ASSUMING NO THERMAL CONDUCTION THROUGH JOINTS NOR RADIANT INTERCHANGE BETWEEN TRUSS MEMBERS. RADIANT INTERCHANGE BETWEEN THE OPAQUE PANELS AND PRIMARY AND SECONDARY TRUSS MEMBERS WAS ACCOUNTED FOR.

Temperatures of Primary Truss Members

TEMPERATURES IN K

ORBIT CONDITION: $\theta = \pi$ rad, FULL ECLIPSE
ECLIPTIC PLANE, GEOSYNCHRONOUS



(a) OUTER FACE MEMBERS

(b) DIAGONAL MEMBERS

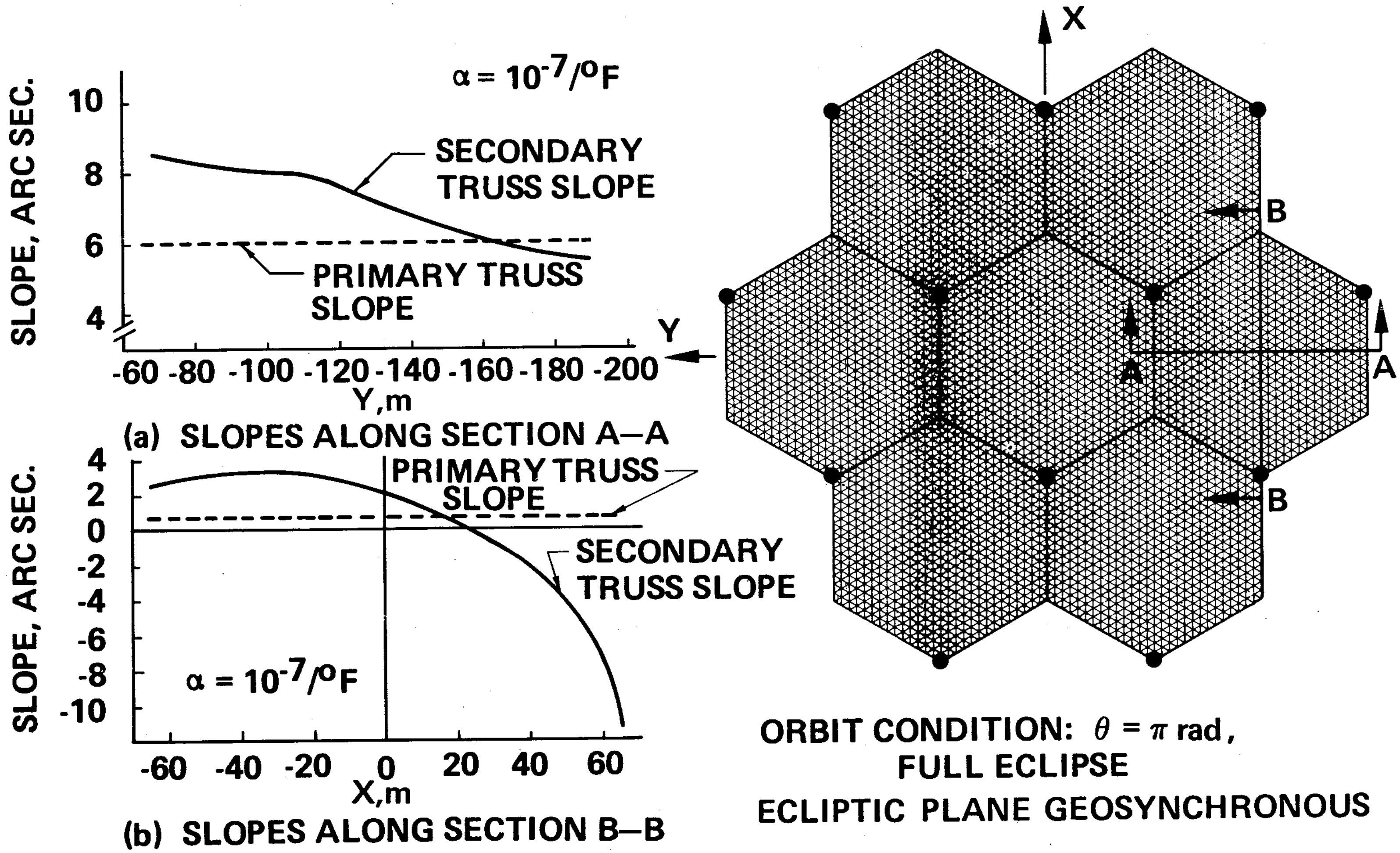
(c) INNER FACE MEMBERS

Figure 12

SLOPES DUE TO THERMAL DISTORTIONS (Figure 13)

PLATFORM SLOPES DUE TO THERMAL DEFLECTIONS WERE CALCULATED USING FINITE-ELEMENT MODELS OF THE PRIMARY AND SECONDARY TRUSSES. FIRST, THERMAL DEFLECTIONS OF THE ATTACHMENT POINTS ON THE PRIMARY TRUSS WERE DETERMINED USING AN EQUIVALENT PLATE ELEMENT MODEL OF THE SECONDARY TRUSS UNITS. THEN A DETAILED FINITE-ELEMENT MODEL OF ONE ARBITRARILY SELECTED SECONDARY TRUSS WAS CONSTRUCTED AND SUBJECTED TO ITS TEMPERATURE DISTRIBUTION AND THE IMPOSED DEFLECTIONS OF ITS THREE ATTACHMENT POINTS DETERMINED FROM THE PRIMARY TRUSS THERMAL DEFLECTION ANALYSIS. THE SLOPES FOR ONE SECONDARY TRUSS AND THE UNDERLYING PRIMARY TRUSS SLOPES ARE SHOWN IN THE FIGURE ALONG THE SECTIONS INDICATED. THE PRIMARY TRUSS SLOPES ARE CONSTANT OVER EACH SECONDARY TRUSS SINCE THE PRIMARY SLOPES ARE DEFINED BY THE LOCATIONS OF THE 3 SUPPORT POINTS FOR EACH SECONDARY TRUSS. EACH SECONDARY TRUSS WILL HAVE SLOPE DISTRIBUTIONS DEPENDENT UPON ITS THERMAL ENVIRONMENT AND THE IMPOSED DEFLECTIONS OF ITS ATTACHMENT POINTS AND IS INDEPENDENT OF THE SLOPE DISTRIBUTION OF THE NEIGHBORING SECONDARY TRUSS. IN GENERAL, THE SLOPES AT THE PERIMETER OF THE TOTAL PLATFORM WILL BE GREATER THAN THOSE IN THE CENTRAL REGION DUE TO GREATER PRIMARY TRUSS DEFLECTIONS AROUND THE PERIMETER.

Slopes Due to Thermal Distortions



ORBIT CONDITION: $\theta = \pi$ rad,
FULL ECLIPSE
ECLIPTIC PLANE GEOSYNCHRONOUS

Figure 13

Summary

- 300m x 300m PLATFORM IS COMPATIBLE WITH STS
  - MULTIFUNCTIONAL MECHANISM OFFERS ADVANTAGES FOR DEPLOYABLE STRUCTURES DESIGN AND TEST PROCEDURES
  - LONG LATTICE COLUMNS ($\geqslant$ 130m) ARE COMPATIBLE WITH STS AND ON-ORBIT ASSEMBLY
- LOW THERMAL DISTORTION OF GRAPHITE COMPOSITE REPEATING MODULE TRUSS STRUCTURE
  - WITHOUT ON-BOARD HEATING - NO APPRECIABLE CURVATURE
  - WITH ON-BOARD HEATING ($\approx$ 3.5 kw/m^2) - LESS THAN ONE ARC MINUTE OF SURFACE DISTORTION

Figure 14

EFFICIENT CONCEPTS FOR LARGE ERECTABLE SPACE STRUCTURES

By

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Paper Presented at

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LARGE SPACE SYSTEMS TECHNOLOGY

January 17-19, 1978

Langley Research Center
Hampton, Virginia

EFFICIENT CONCEPTS FOR LARGE ERECTABLE SPACE STRUCTURES

(Figure 1)

The development of credible concepts for large space structures requires design information on structures of unprecedented proportions. The Langley Research Center has initiated studies of basic generic concepts for large space structural elements in order to provide meaningful standards for evaluating competing types of construction, to ensure that the unique behavior aspects of large lightweight members are not overlooked in design, and to provide experimental and analytical data on promising concepts.

In the present paper, the status of LaRC development of the nestable column concept will be reviewed including results of member and truss component tests, and planned assembly studies. In addition, more recent studies of alternative member concepts will be presented. Preliminary results on relative efficiency of several types of truss-type columns will be compared and future test plans will be discussed.

EFFICIENT CONCEPTS FOR LARGE ERECTABLE SPACE STRUCTURES

<u>PURPOSE OF GENERIC CONCEPT STUDIES</u>

- ESTABLISH STANDARDS TO COMPARE COMPETING CONSTRUCTIONS
- CONFIRM METHODS OF DESIGN FOR LARGE, LIGHTWEIGHT MEMBERS
- PROVIDE DATA ON PROMISING CONCEPTS

<u>STATUS OF LaRC STUDIES</u>

NESTABLE COLUMNS

- MEMBER TESTS
- TRUSS COMPONENT TESTS
- ASSEMBLY STUDIES

ALTERNATE CONCEPTS

- EFFICIENCY STUDIES
- FUTURE TESTS

Figure 1

NESTABLE COLUMN CONCEPT

(Figure 2)

The nestable tapered column concept combines a structurally efficient wall configuration with a simple concept to achieve high packaging efficiency. The wall configuration is a hollow tapered tubular section made largely from unidirectional graphite to give the column low axial expansion and good mass/buckling strength characteristics for compression loadings. High packing density is achieved by using half-column tapered sections and nesting them like paper cups for transportation into orbit. Previous studies (ref. 1 and 2) have shown that cost-efficient weight critical payloads can be achieved in the Space Shuttle Transportation System. On orbit, the columns may be assembled and used as building blocks in large space truss structures (e.g. ref. 3).

The figure illustrates some five-meter graphite-epoxy columns with aluminum center and end joints. The elements were designed and constructed to evaluate the practicality of lightly loaded (1000 lb.) members.

REFERENCES

1. Bush, H. G.; and Mikulas, M. M., Jr.: A Nestable Tapered Column Concept for Large Space Structures. NASA TMX-73927. July, 1976.
2. Bush, H. G.; and Mikulas, M. M., Jr.: Some Design Considerations for Large Space Structures. AIAA Paper No. 77-395, March 21-23, 1977.
3. Mikulas, M. M., Jr.; Bush, H. G.; and Card, M. F.: Structural Stiffness, Strength and Dynamic Characteristics of Large Tetrahedral Space Truss Structures. NASA TMX-74001, March, 1977.

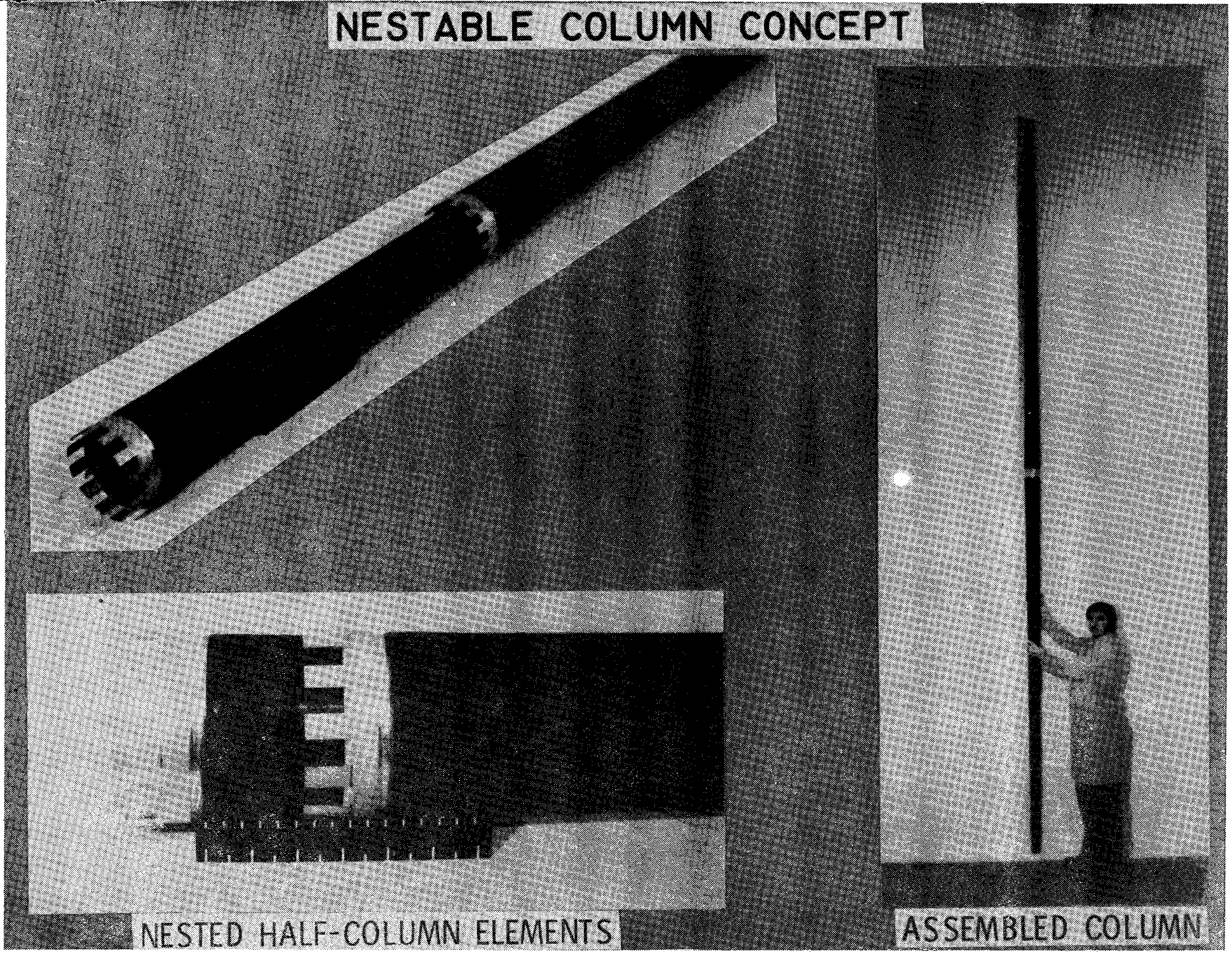
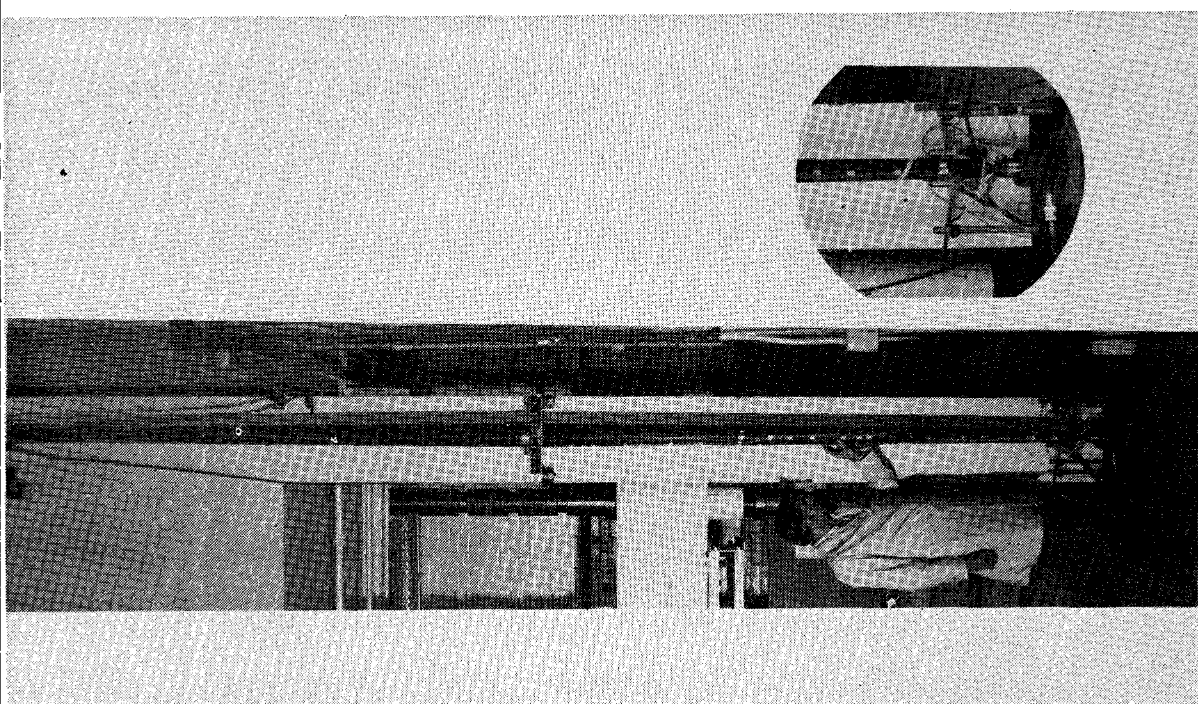

NESTABLE COLUMN CONCEPT
NESTED HALF-COLUMN ELEMENTS
ASSEMBLED COLUMN

Figure 2

5-METER NESTABLE COLUMN BUCKLING TESTS

(Figure 3)

In order to verify the behavior of long slender nestable columns, a series of column buckling tests were conducted with 5.2-meter-long graphite-epoxy specimens mounted vertically as shown in the figure. Simple support end conditions were achieved by the use of ball-and-cup end fittings. Of interest in the tests was low-strain material behavior ($\varepsilon < .0005$), repeatibility of tests, and behavior of the interlocking leaf-spring center joint under load.



5 METER NESTABLE COLUMN BUCKLING TEST

Figure 3

GRAPHITE NESTABLE COLUMN BUCKLING TEST RESULTS
(Figure 4)

As summarized in the Table, seven columns were tested. As indicated, the column walls were nearly unidirectional (0° fibers alined with loading direction) with 90° surface plies. Also shown on the Table are values of measured deviation from straightness in terms of the amplitude of maximum lateral deviations δ and the column length ℓ. The low values of both the initial imperfection amplitudes and buckling load permitted the columns to be buckled repeatedly without damage with repeatable test results.

In comparing test buckling results with analysis, it was discovered that some test articles were made with overlaps in 0° plies causing significant increases in thickness. The changes from nominal extensional stiffness are shown in the Table. By correcting thicknesses of the analysis model, excellent agreement was obtained between experimental buckling loads $P_{cr_{exp}}$ and theoretical loads $P_{cr_{th}}$ based on Timoshenko tapered column theory.

GRAPHITE NESTABLE COLUMN
BUCKLING TEST RESULTS

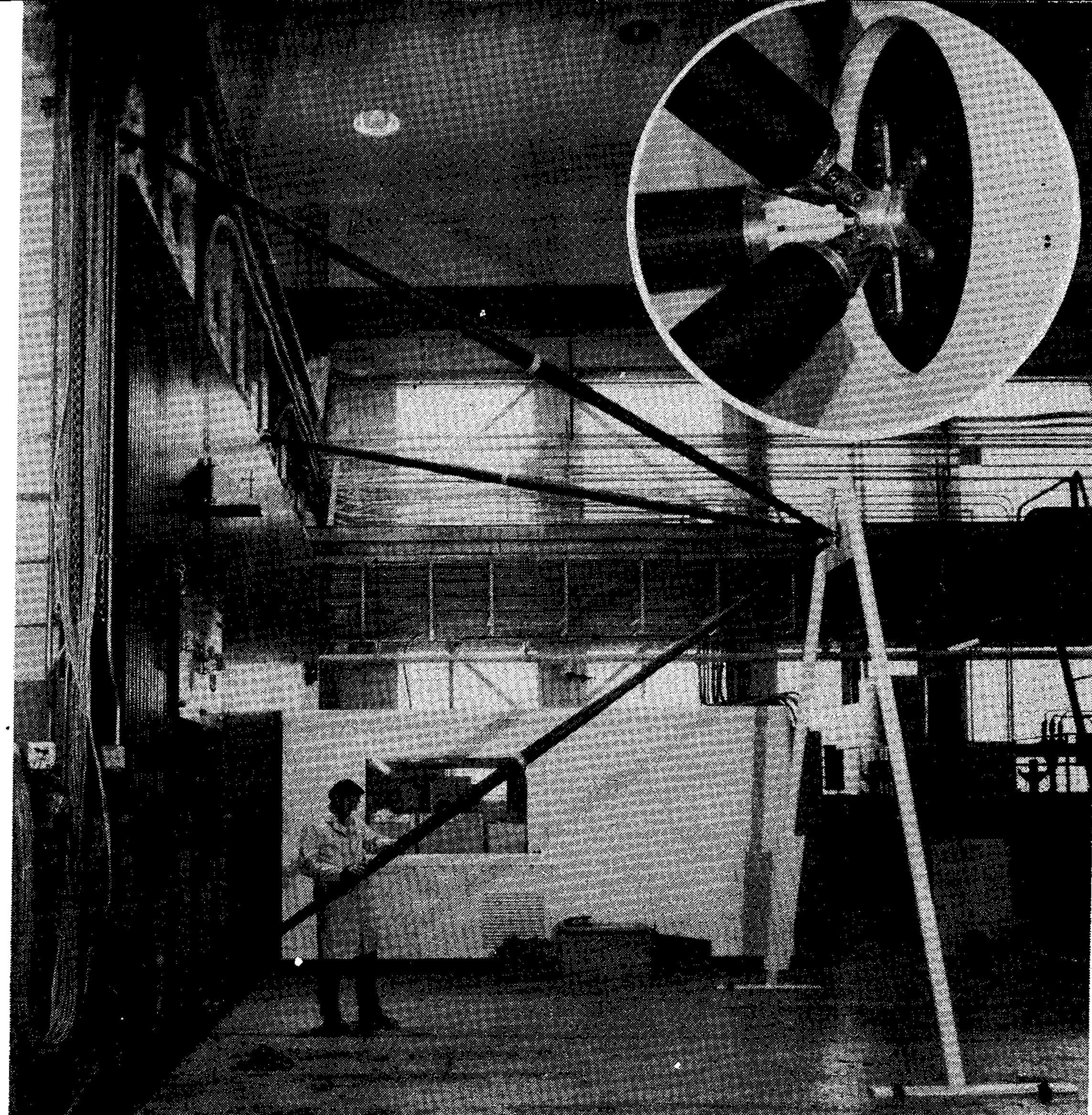
TEST SPECIMEN	LAYUP	$\frac{\delta}{L}$	$\frac{(Et)_{exp.}}{(Et)_{nom.}}$	$\frac{P_{exp.}}{{}^{*}P_{th.}}$
1	$90/0_3/90$	.0009	.939	.990
2	$90/0_4/90$	0	.964	.988
3		-	1.074	.996
4		-	1.063	1.000
5		.0004	1.088	1.003
6		.0007	1.074	.997
7		.0004	1.075	1.017

* TIMOSHENKO TAPERED COLUMN WITH CORRECTED STIFFNESS

Figure 4

TRIPOD BUCKLING TESTS
(Figure 5)

In addition to the column tests, three of the tubes were assembled into a tripod as shown in the photograph. Loads were applied to the apex of the tripod in a plane parallel to the base. Of interest in these tests was the behavior of the aluminum cluster joint at the tripod apex.



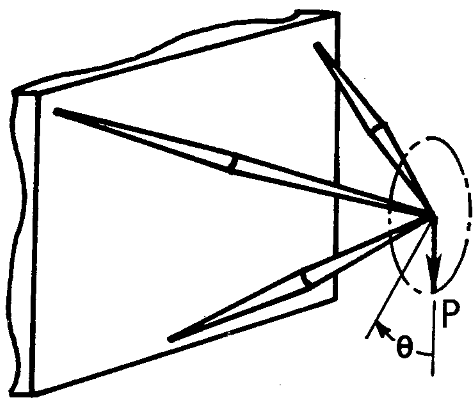
TRIPOD BUCKLING TEST

Figure 5

TRIPOD BUCKLING TEST RESULTS
(Figure 6)

Buckling results are summarized in the Table for three values of the loading angle Θ. A series of buckling calculations with various boundary conditions were made using a finite element code (SPAR). As suggested, test results are bounded by simple support and simple support-clamped boundary conditions. Using the test data for $\Theta = 0°$ and previous test data for simply supported columns, buckling loads were estimated to be about 1.27 times loads obtained from the tripod analysis with simply supported boundaries. As the Table indicates, this method gives reasonable estimates for $\Theta = 30°$ and $\Theta = 60°$ test results as well.

TRIPOD BUCKLING TEST RESULTS



$$\frac{P_{exp}}{P^{*}_{th}}$$

θ^{o}	SIMPLE SUPPORT CLUSTER AND BASE	SIMPLE SUPPORT CLUSTER, CLAMPED BASE	ELASTIC** BOUNDARIES
0	1.309	.837	1.035
30	1.342	.858	1.061
60	1.171	.749	.926

*SPAR FINITE ELEMENT MODEL (12 ELEMENT BEAMS)

**ELASTIC SUPPORT FACTOR OF 1.27 X SIMPLE SUPPORT (ESTIMATED FROM COLUMN TESTS AND $\theta = 0^{o}$ TEST RESULTS)

Figure 6

VIBRATION TEST RESULTS

(Figure 7)

Limited surveys were made of the vibration behavior of the columns and the assembled tripods using a low force shaker and a lightweight movable accelerometer. A 23 kg tip mass was attached to the tripod cluster joint to help discriminate between individual tripod member frequencies. In the Table, values of the lower bending frequencies are compared with theoretical predictions. Column test results were compared with a shell-of-revolution analysis which accounted for aluminum end fittings and abrupt changes in thickness. Experimental results were within ± 5% of theoretical predictions. Tripod test results were compared with finite-element predictions (SPAR) using the same model as used for buckling. The model employed simply supported base supports and a simply supported apex with a point tip mass. Experimental results deviated by a maximum of 8% from theory.

VIBRATION TEST RESULTS

COLUMN TESTS

TEST SPECIMEN	BOUNDARY CONDITIONS	MODE	$\frac{f_{exp}}{f_{th}^{*}}$
1	FREE - FREE	1ST BENDING 2ND BENDING 3RD BENDING	.95 .95 .95
1	SIMPLY SUPPORTED	1ST BENDING 2ND BENDING	.97 .96
2	SIMPLY SUPPORTED	1ST BENDING 2ND BENDING	1.04 1.05

TRIPOD TESTS

5 6 7	TRIPOD WITH BASE FIXTURES, CLUSTER JOINT AND TIP MASS	1ST BENDING 1ST BENDING 1ST BENDING	.92 .93 .93

* SHELL OF REVOLUTION ANALYSIS FOR COLUMNS; SPAR FINITE ELEMENT ANALYSIS FOR TRIPOD

Figure 7

36-ELEMENT OCTETRUSS TEST

(Figure 8)

As part of further evaluation of nestable columns as components of large space trusses, the multi-element truss section shown is scheduled for tests at LaRC this summer. The section shown is a small representative module of the so-called tetrahedron truss or octetruss which has been investigated in previous studies. As suggested, the truss will be cantilevered from a sturdy backstop and loaded with a tipload. Of particular interest here, is the behavior of the 9-element joint shown in the top face of the truss. The truss will be subjected to bending load to induce column buckling and a vibration survey will be conducted.

36–ELEMENT OCTETRUSS TEST

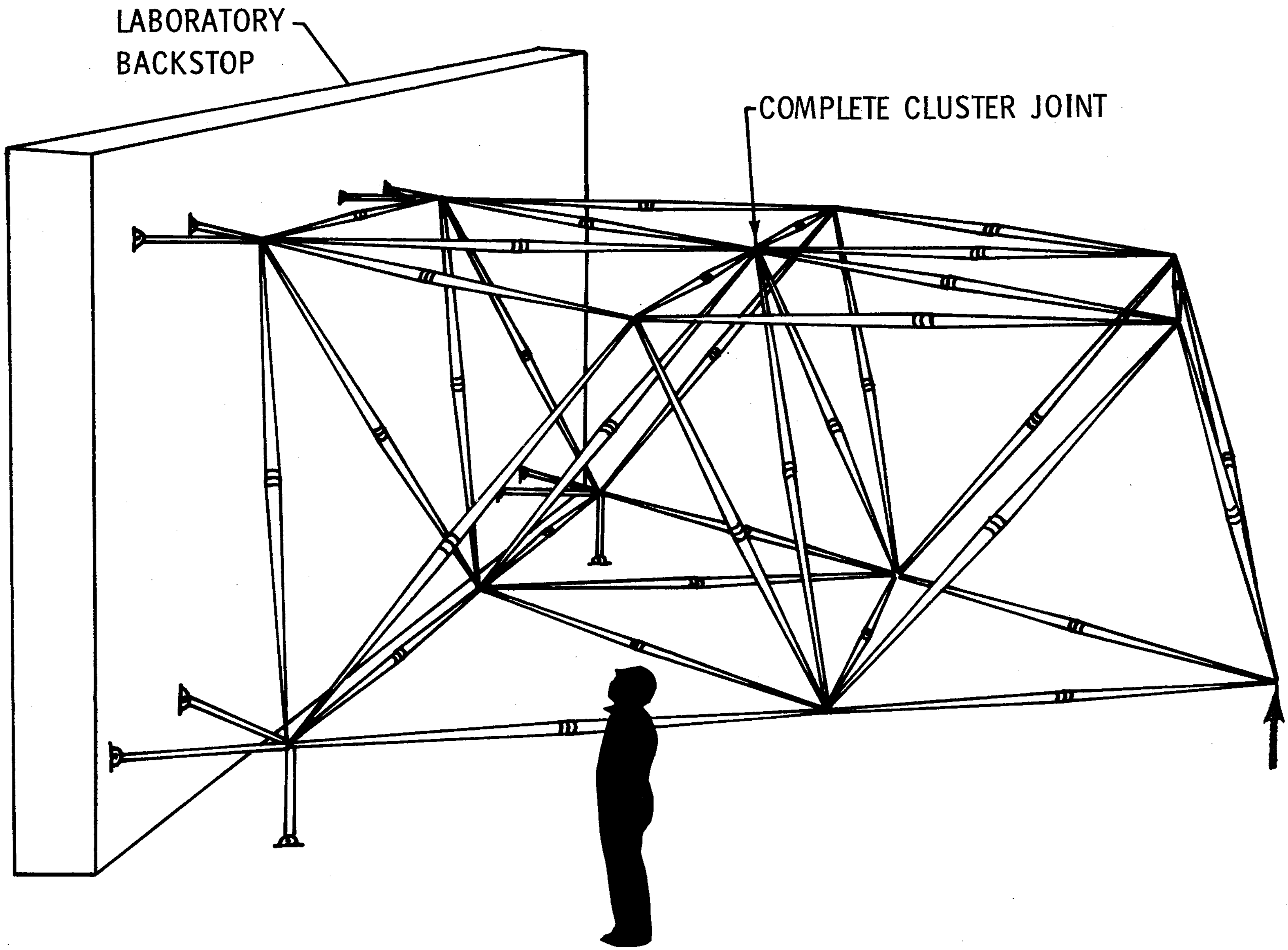
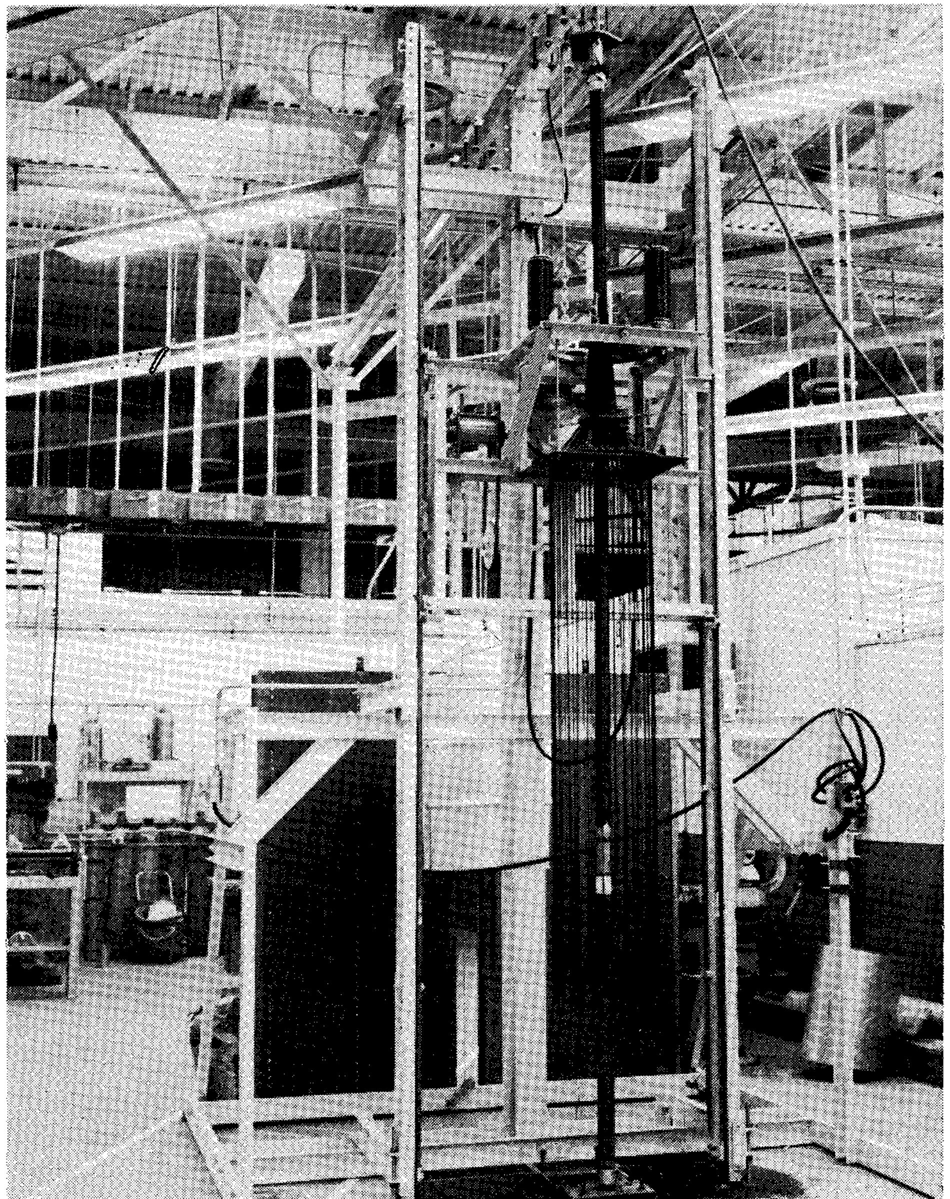


Figure 8

MANUFACTURING FACILITY FOR OCTETRUSS ELEMENTS
(Figure 9)

As part of efforts to reduce costs of nestable column truss members, a manufacturing facility is being developed by Lockheed. The photo shows a tube-stand facility suitable for making half columns up to 5 meters in length. The heart of the facility is a movable platform on which is mounted spools (near top of photo) for circumferential winding and a gathering ring and tension plate for laying 0° degree filaments. The tension plate is perforated with small guide holes containing ceramic rollers which can be tensioned against the filament.

To initiate the tube manufacturing of a 90-0-90 configuration, a tapered mandrel is first placed in the stand. Then a single upward winding pass from bottom to top of the mandrel is made to form the inner surface circumferential ply. The tube is then completed with a single downward pass of the movable platform in which the tensioned 0 degree filaments are fed through the tension plate and gathering ring, then overwound with the external 90° ply. Either film or painted resin can be used in the process and tubes may be bagged and cured in place.



MANUFACTURING FACILITY FOR OCTETRUSS ELEMENTS

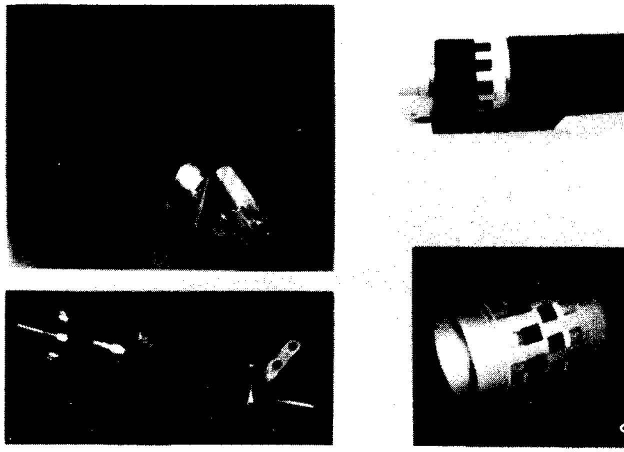
Figure 9

ASSEMBLY TECHNOLOGY
(Figure 10)

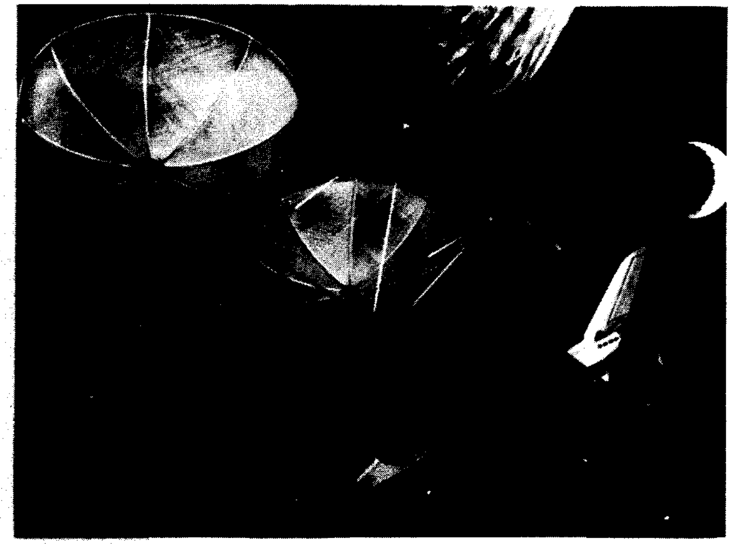
The final step in the development of the nestable column concept is the invention of a credible automated assembly process. Our studies of joint concepts suggest that assembly concepts will strongly dictate the most suitable center and end joints for a member. Preliminary studies have been performed by Rockwell International on the feasibility of using a modified shuttle Remote Manipulator System to erect a large antenna platform using nestable columns. More recently Lockheed has suggested that automatic erector system based on a lightweight translating parallelogram fixture might be feasible. Preliminary estimates of space truss assembly times suggest a reasonable potential for the automated erector, but there are great uncertainties in the realism of assembly time predictions.

It appears that there is a serious need for in-depth investigations of competing assembly concepts. In view of the range of complexities and costs of such systems, it would seem prudent to first establish baseline <u>manned</u> assembly timelines for early shuttle assembly activities. Furthermore, assembly times of automated devices should be established by laboratory tests of scaled erectors. In Phase One of the Large Space Systems Technology Program, such in-depth studies are planned.

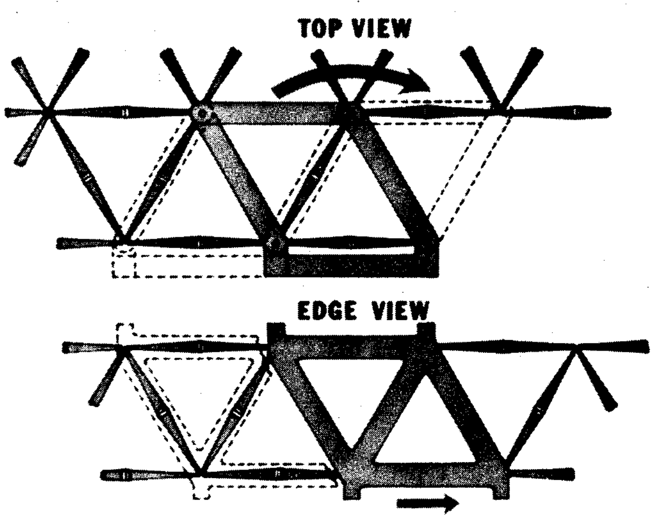
ASSEMBLY TECHNOLOGY



JOINT CONCEPTS



SHUTTLE - RMS ASSEMBLY



AUTOMATIC ERECTOR CONCEPT

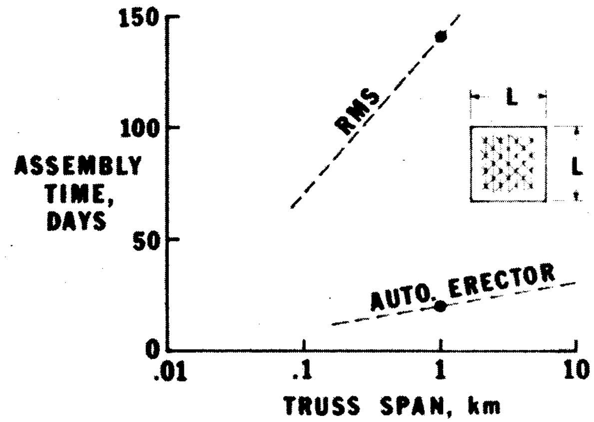


Figure 10

DESIGN STUDIES ON ALTERNATE CONCEPTS
(Figure 11)

The chart shows the three main aspects of LaRC investigations of alternative concepts to the nestable column. In the design of very long, lightly loaded column members, concepts such as the tri-element truss column and columns with central compression posts stabilized with pretensioned wires look promising. Such concepts are effective because they overcome structural efficiency limitations posed by minimum gauge solid-skin structures. Generic studies of the relative weights and efficient proportions of such members are underway.

An aspect that is under investigation is a method of simplifying relative efficiency comparisons in regions where minimum gauge structure is predominant. Classical efficiency comparisons have used the mass parameters $\frac{W}{L^3}$ and a loading index $\frac{P}{L^2}$. In the minimum gauge regime, families of curves of constant length are awkward to compare. An example of a simplified efficiency chart will be discussed subsequently.

A final aspect of the LaRC design studies is the effect of initial imperfections (i.e. deviation from straightness) on column structural buckling strength. Recent work suggests that highly optimized structures are particularly sensitive to relatively small amplitude imperfections.

DESIGN STUDIES ON ALTERNATE CONCEPTS

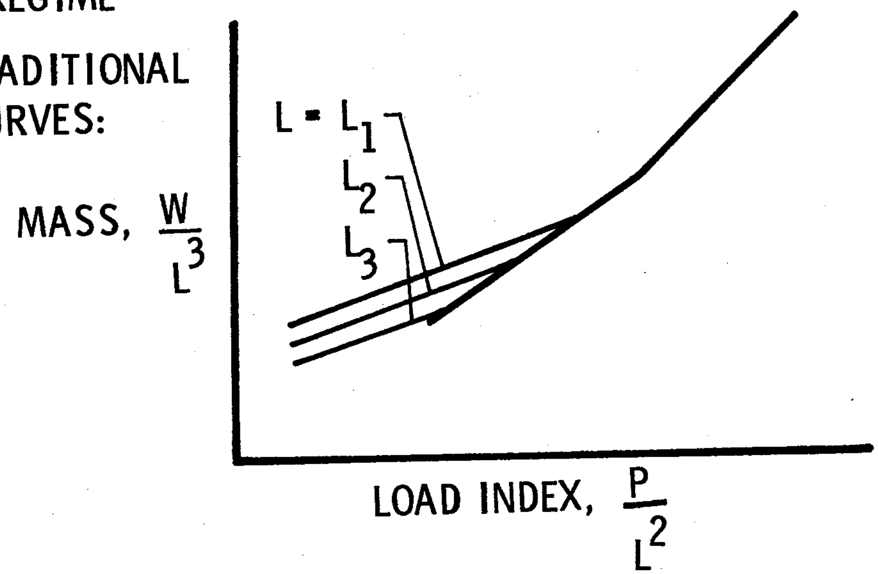
- STUDYING CONFIGURATIONS MORE APPROPRIATE FOR ULTRA LARGE STRUCTURES

  (1) TRI - ELEMENT TRUSS - COLUMNS

  (2) TENSION - STABILIZED COLUMNS

- REDEFINING MASS/LOADING INDEXES TO SIMPLIFY MASS ESTIMATES IN LIGHTLY LOADED REGIME

TRADITIONAL CURVES:



- INVESTIGATING EFFECTS OF INITIAL IMPERFFCTIONS

Figure 11

MASS-STRENGTH CHARACTERISTICS OF GRAPHITE-EPOXY COMPRESSION COLUMNS

(Figure 12)

The figure shows a preliminary comparison of the relative weights of three types of graphite-epoxy column construction. The parameters are column mass normalized as $\frac{W}{\ell^{5/3}}$ (W = weight, ℓ = length) and the applied compressive load P. Efficiency curves are shown for; 1) minimum gauge graphite hollow tubular construction (t = .015in.) (representative of the nestable column construction ignoring the effects of taper), 2) tri-element truss construction with solid longerons and 3) tri-element truss construction with hollow tubing longerons. As expected, the curves shown illustrate the superiority of open-construction over solid tubing for the lower loads.

For tri-element truss construction with solid tube longerons, a representative design point for a 50m foldable astromast is shown and is in agreement with the computed curve. A data point for a General Dynamics design of a manufactured-in-space tri-element truss beam is also shown; it is somewhat higher than predicted curves because of open rather than closed construction of the column longerons. Finally a data point is shown for a Boeing design for Solar Satellite Power Station member. The point illustrates that gigantic columns with longerons made from nestable column subelements possess high efficiency.

MASS/STRENGTH CURVES FOR Gr/E COMPRESSION COLUMNS

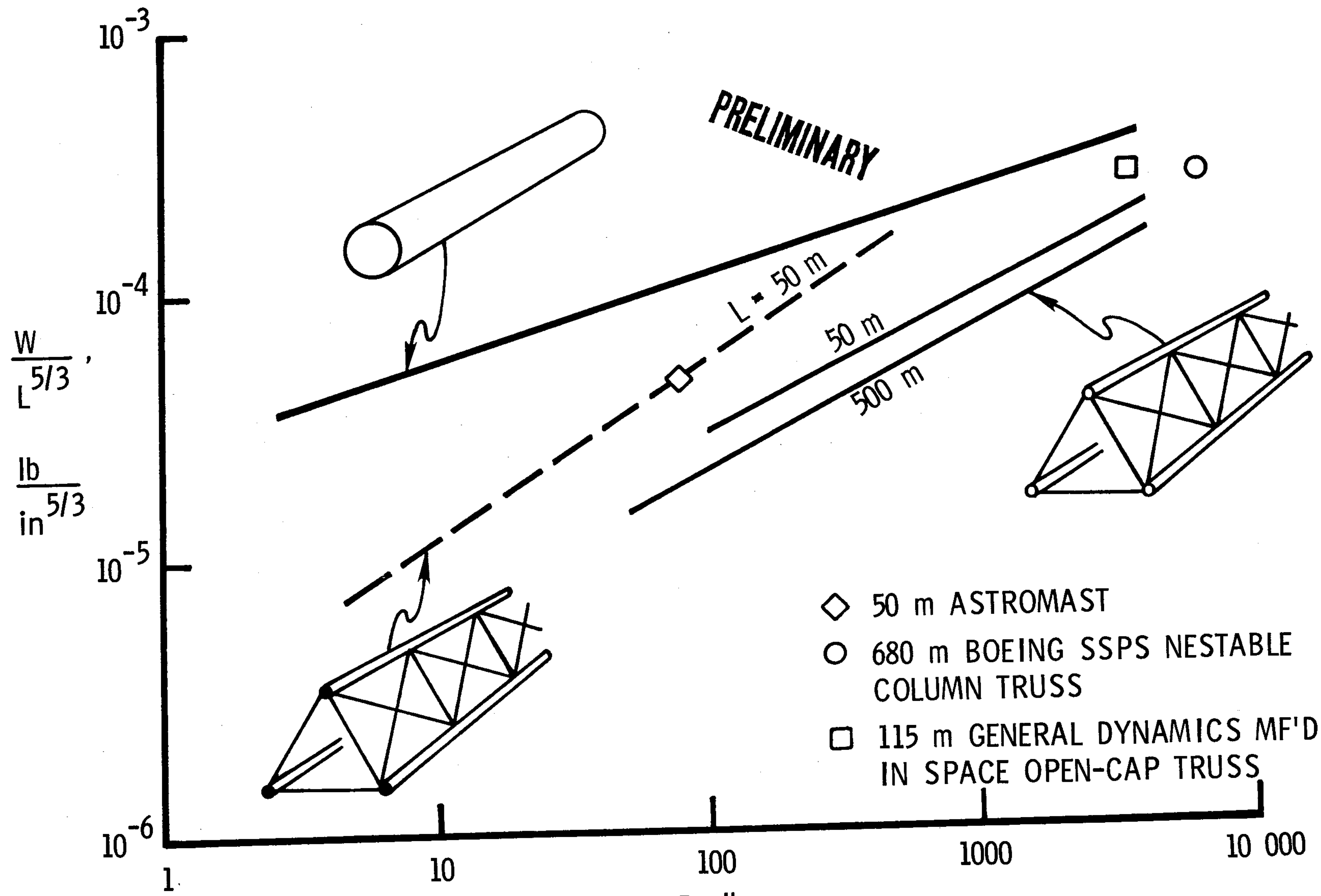


Figure 12

EFFECTS OF INITIAL IMPERFECTIONS
(Figure 13)

Although tri-element truss columns look very attractive from a mass-efficiency point of view, preliminary studies suggest that the members are sensitive to the effects of imperfections. The left-hand curve shows an example of the increase in mass needed to make an imperfect column carry the same load as a perfectly straight member. A typical recommended design value is shown for the imperfection amplitude $\frac{\delta}{\ell}$ as well as the values obtained from tests of the 5 meter nestable columns. If the recommended value is used, a factor of two in weight increase is required to develop the same load carried by a perfectly straight member. Also shown are values of imperfection measured on the nestable columns which suggest the recommended design value (developed in 1930's) could be overly conservative.

A second difficulty with columns with large imperfections is suggested in the right-hand figure. For the recommended imperfection design value, a nondimensionalized load-shortening curve is shown. The curve indicates that for possible operational loads in the column (say limit load with a safety factor of 1.5, $\frac{P}{P_E} = .67$), large nonlinear behavior is evident.

EFFECTS OF INITIAL IMPERFECTIONS

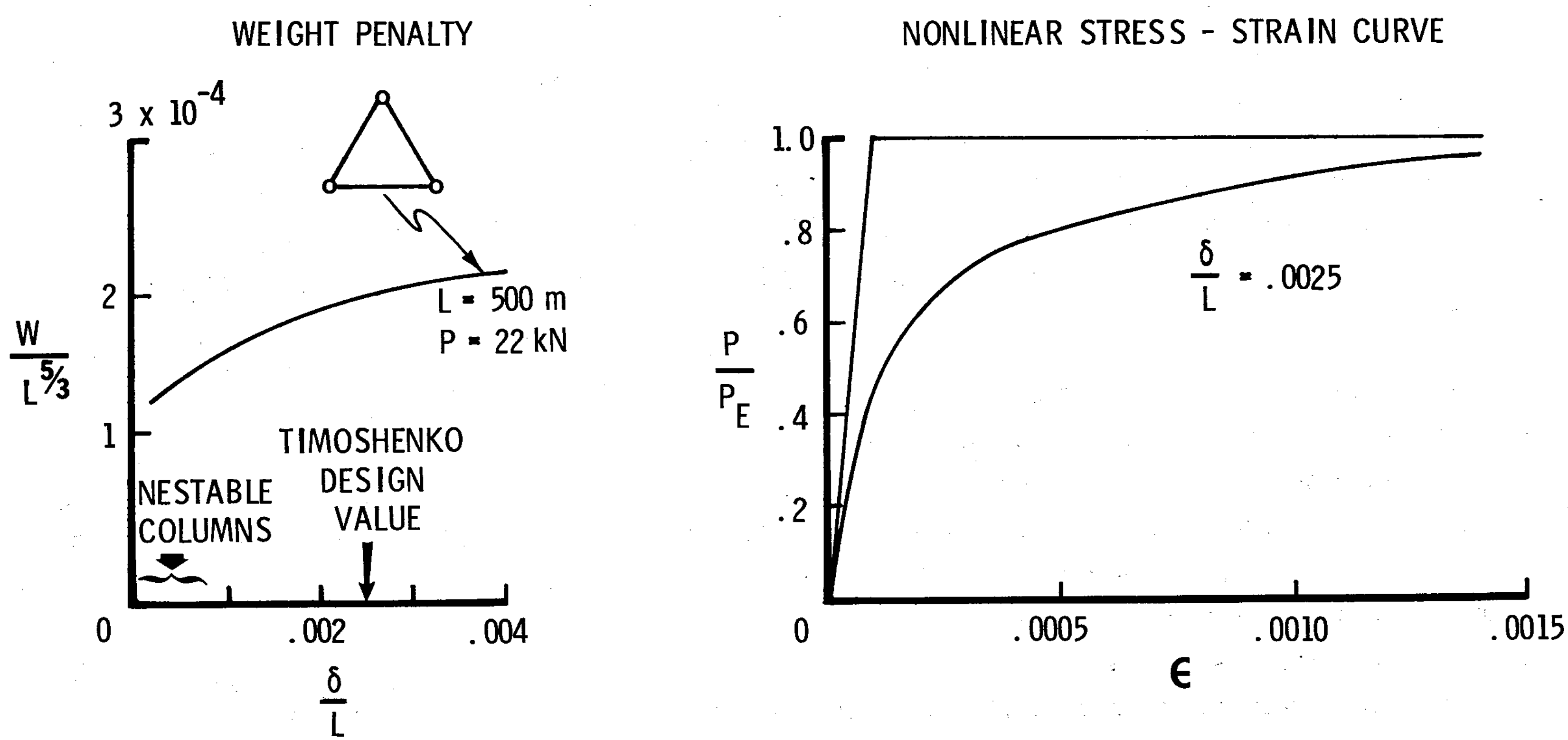


Figure 13

LATTICE COLUMN TESTS IN STRUCTURAL DYNAMICS RESEARCH LABORATORY

(Figure 14)

In order to obtain experimental data on the strength and imperfection behavior of very large members, a test facility is being developed in the Langley Dynamics Research Laboratory. In this building, a large tower capable of accomodating members up to 25 meters in length will be modified for vibration and buckling tests as suggested in the figure. A short section of a sturdy aluminum lattice (tri-element truss) column has been designed and tested to verify detail design. A 24-meter column is presently under construction at Langley and should be ready for test in late spring. Plans call for a series of tests of at least two types of competing construction.

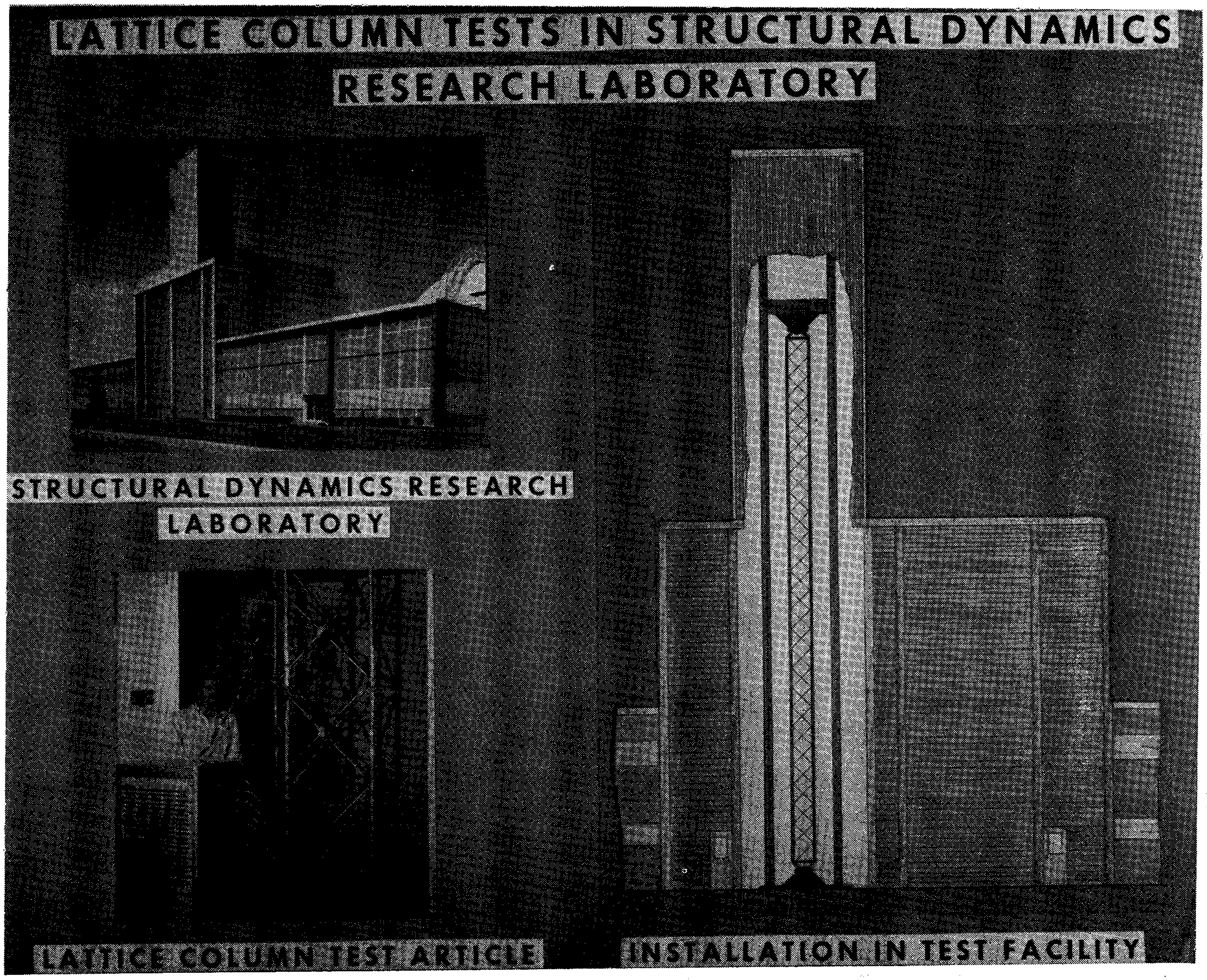

LATTICE COLUMN TESTS IN STRUCTURAL DYNAMICS RESEARCH LABORATORY
STRUCTURAL DYNAMICS RESEARCH LABORATORY
LATTICE COLUMN TEST ARTICLE
INSTALLATION IN TEST FACILITY

Figure 14

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Efficient Concepts for Large Erectable Space Structures

1. Selection of a 1000 Pound Loading Condition

The selection of 1000 pounds represents a reasonable practical level compatible with a range of design considerations. The loads predicted for any individual member can vary widely depending upon assumptions which define the control systems.

2. Assembly Considerations for Nestable Columns

The next phase of development for nestable columns includes methods for assembly. Specific studies have included utilization of the Shuttle manipulator and a lightweight unit proposed for combined assembly and erection. There are studies underway toward evaluating the functional concepts (credibility) and construction times. As part of this evaluation there is some agreement on the need to establish baseline data relative to EVA performed assembly operations.

SPACE FABRICATION & ASSEMBLY OF GRAPHITE COMPOSITE TRUSSES

D. J. Powell

Presented at
Government/Industry Seminar on
Large Space Systems Technology

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GENERAL DYNAMICS
Convair Division

17018000J2051

BASELINE SYSTEM CONCEPT (Figure 1)

A baseline LSS concept is shown in the facing illustration. In mid-1982, fabrication/assembly systems and prepackaged raw materials are delivered by Shuttle to a 300 nautical mile (556 Km) circular orbit.

Upon system deployment from the stowed position, a beam-builder, moving to successive positions along a Shuttle-attached assembly jig, automatically fabricates four triangular beams, each 200 meters long. Retention of the completed beams is provided by the assembly jig.

The beam-builder then moves to the position shown and fabricates the first of nine shorter, but otherwise identical, cross-beams. After cross-beam attachment, the partially completed assembly is automatically transported across the face of the assembly jig to the next cross-beam location, where another cross-beam is fabricated and installed. This process repeats until the "ladder" platform assembly is complete. The cross-beam installation process represents an opportunity for development/evaluation of EVA-assisted/partially automated techniques as shown.

Upon platform assembly completion, both structural and thermal response tests are conducted and RMS/platform release/recapture techniques are developed, thus completing the seven-day mission cycle. Soon after, a revisit mission installs experimental and functional subsystem equipment for an initial application and further testing.

BASELINE SYSTEM CONCEPT

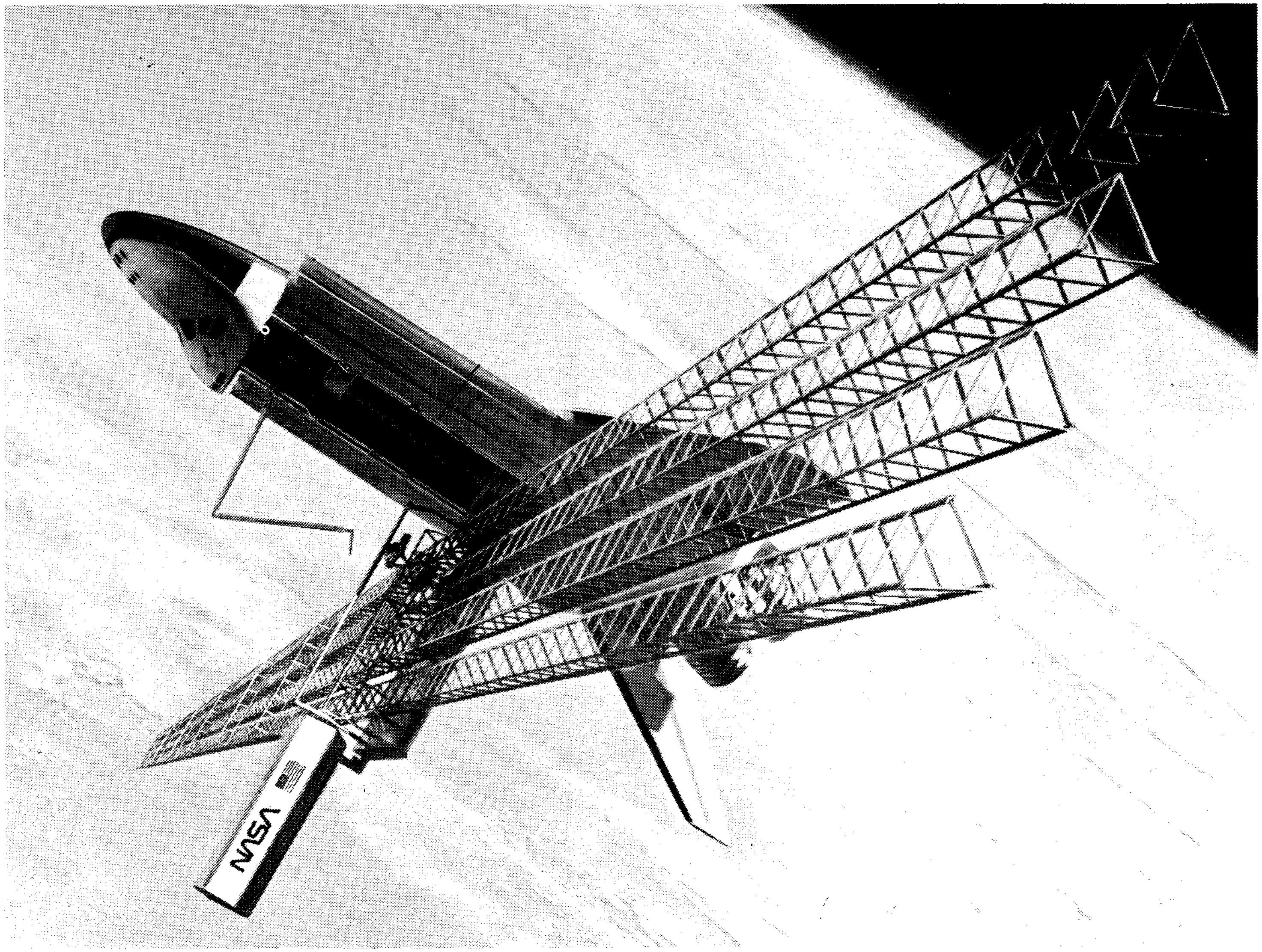

Figure 1

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MATERIALS AND PROCESSES REQUIREMENTS (Figure 2)

The structural elements which we wish to fabricate on-orbit for eventual use in large space structures applications must be very light and adequately rigid. In addition, the material from which they are made when of the graphite thermoplastic type must be formable and assemblable into appropriate structural elements on orbit. The material is carried to orbit compactly packaged on reels, formed into structural sections of indefinite length and assembled into workable trusses. The use of graphite thermoplastic materials enables the designer to take advantage of the designed low coefficient of thermal expansion. Because of the eventual extensive use forseen for this application, it is also necessary that the material should be of low cost. Pitch fiber graphite is expected to have this characteristic during the time frame under consideration, possibly due to extensive use of the material in automotive weight-saving applications. For operational reasons, it is necessary to minimize the amount of energy needed to do the forming of the structural members. This is particularly true for applications depending on the Shuttle power supplies. It is also necessary that the processes do no harm to the material during forming so that full material properties are available to the designer. The beam fabrication process should also produce elements which are as straight and twist-free as possible. The requirements in these latter areas have yet to be formally established.

MATERIALS & PROCESSES REQUIREMENTS

MATERIALS

- FORMABLE & ASSEMBLABLE ON ORBIT
- LOW COEFFICIENT OF THERMAL EXPANSION
- LOWEST COST FOR EVENTUAL LARGE SCALE USE

PROCESSES

- MINIMUM ENERGY NEED
- NO HARM TO MATERIAL
- STRAIGHT, TWIST-FREE BEAM

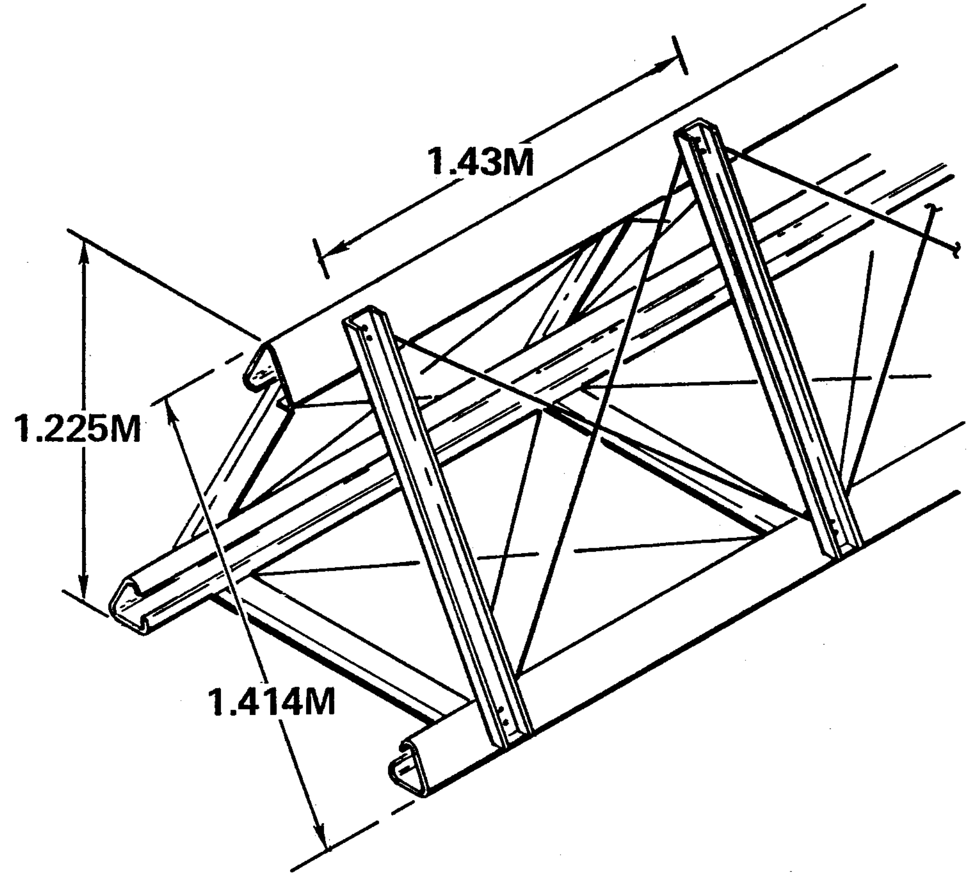

Figure 2

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LAMINATE CROSS SECTION (Figure 3)

Three forms of laminate design are under consideration for large space structures use. All three forms have the promise of doing the job efficiently. Each has pros and cons.

1. Sandwiched 0° Tape

   This concept uses several plies 0° pitch graphite sandwiched between single plies of 104 or 120 glass cloth. The number of 0° plies and the thickness of cloth are determined by structural requirements (loads and stiffness).

2. Woven Fabric

   This concept uses a single ply of a fabric woven such that 0°, 90°, and ± 45° fibers are present; the amounts of each are dependent on the structural requirements. The fabric would necessarily be thicker than that presently used in aerospace applications and would have to be specifically designed to meet space platform or other specific load requirements.

3. Multi-ply Laminates

   Multi-ply laminates are the traditional form of laminate used in composite structures. They consist of several layers of 0° graphite plies orientated in the directions best suited to meet the requirements of a particular structural element. As such, they can be optimized for each application.

For near term development, the most flexible, least cost, and least risk approach appears to be the sandwiched 0° tape. A typical generic laminate which is weight effective is $(120/0_2/120)_T$ glass/VSB-32T graphite/glass in a polysulfone (P-1700) matrix. HMS fiber could be considered if VSB-32T pitch fiber in polysulfone is not available for early development. The chart illustrates the laminate construction. For longer term development, more investigation of a tailored, single-ply fabric should be conducted.

LAMINATE CROSS SECTION

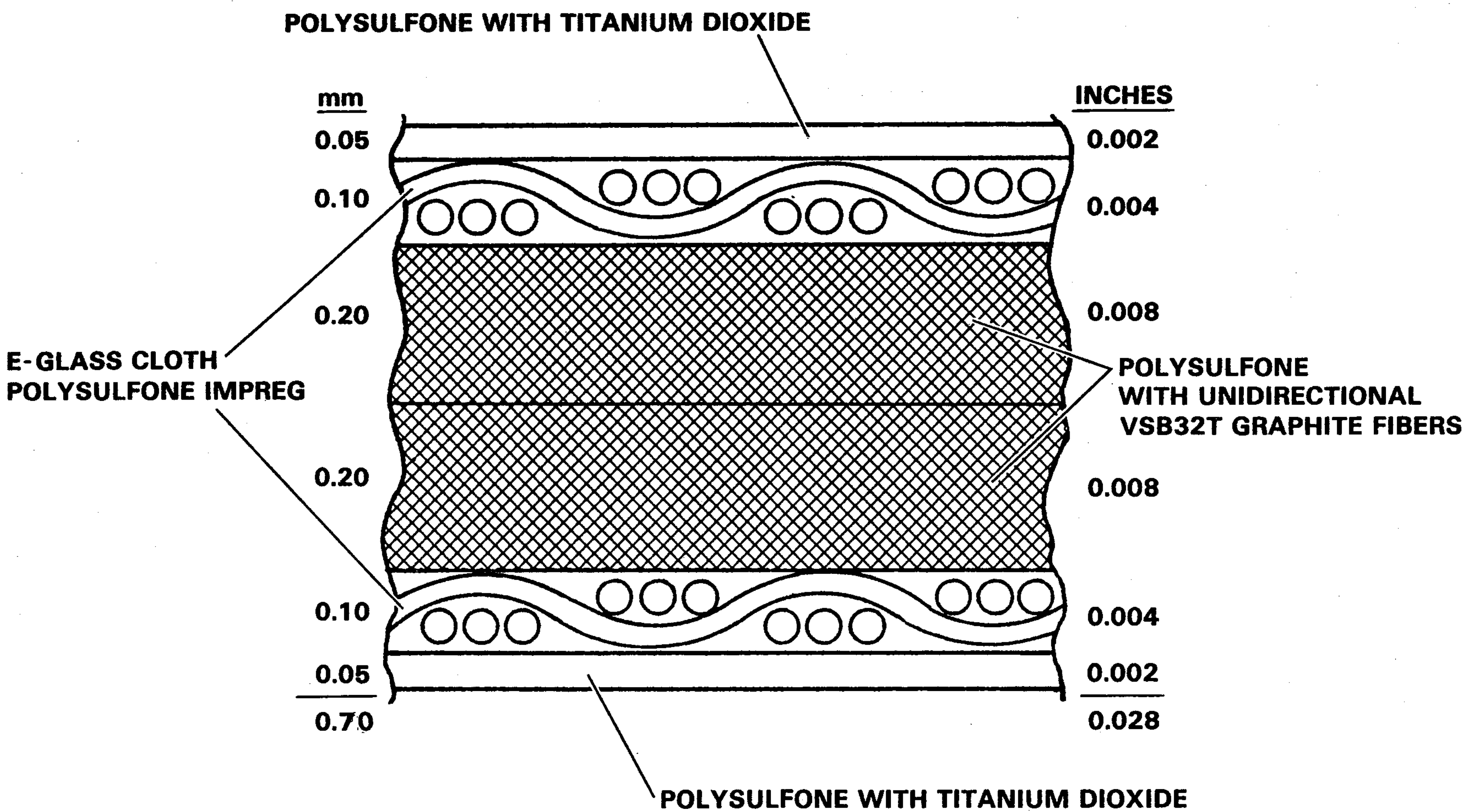

Figure 3

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CAP SECTION ROLLTRUSION HEAD (Figure 4)

The machine element for forming the cap section consists of a long box frame with the reel for the laminate strip mounted at one end and the drive for pulling the laminate through the machine mounted at the other end. The reel (Section A-A opposite) has an adjustable drag brake on one hub and is hooded to contain the "clockspring" tendency of the laminate in the event of drag brake malfunction. A set of rollers that maintains a minimum bend radius of 300 mm from full to empty reel is located on the frame adjacent to the reel. Next to the rollers is the heating module (Section B-B) which is one beam bay (1434 mm) in length. The heat is confined to the bend zones in the laminate by using linear resistance heaters in parabolic reflectors.

Coolant flowing through four passages in each heater block carries away the small amount of heat absorbed in the reflector ($\approx 6\%$).

Temperature sensors are installed on the opposite side of the laminate to provide feedback to the temperature control system. Coolant flowing through two passages provides thermal control of each sensor to assure a stabilized temperature reference.

A forming module consisting of a series of rollers and guideshoes interspersed with heaters is located immediately downstream of the heating module. The components in this module are placed on a curvature that has been carefully developed to coincide with the natural flow of the laminate during the forming process to avoid any wrinkling/malformation in the plastic bend zones that would damage the graphite fibers in the laminate. The laminate is first formed to a Vee shape, then the flanges are formed. Heaters and temperature sensors are interspersed between each stage. The heaters in the heating and forming sections are controlled to provide a bend zone temperature of $425 \pm 25°$ F in the forming section. The plastic bend zones are protected from scuffing by relief cuts in the guide shoes and teflon coating on the idler rollers.

CAP SECTION ROLLTRUSION HEAD

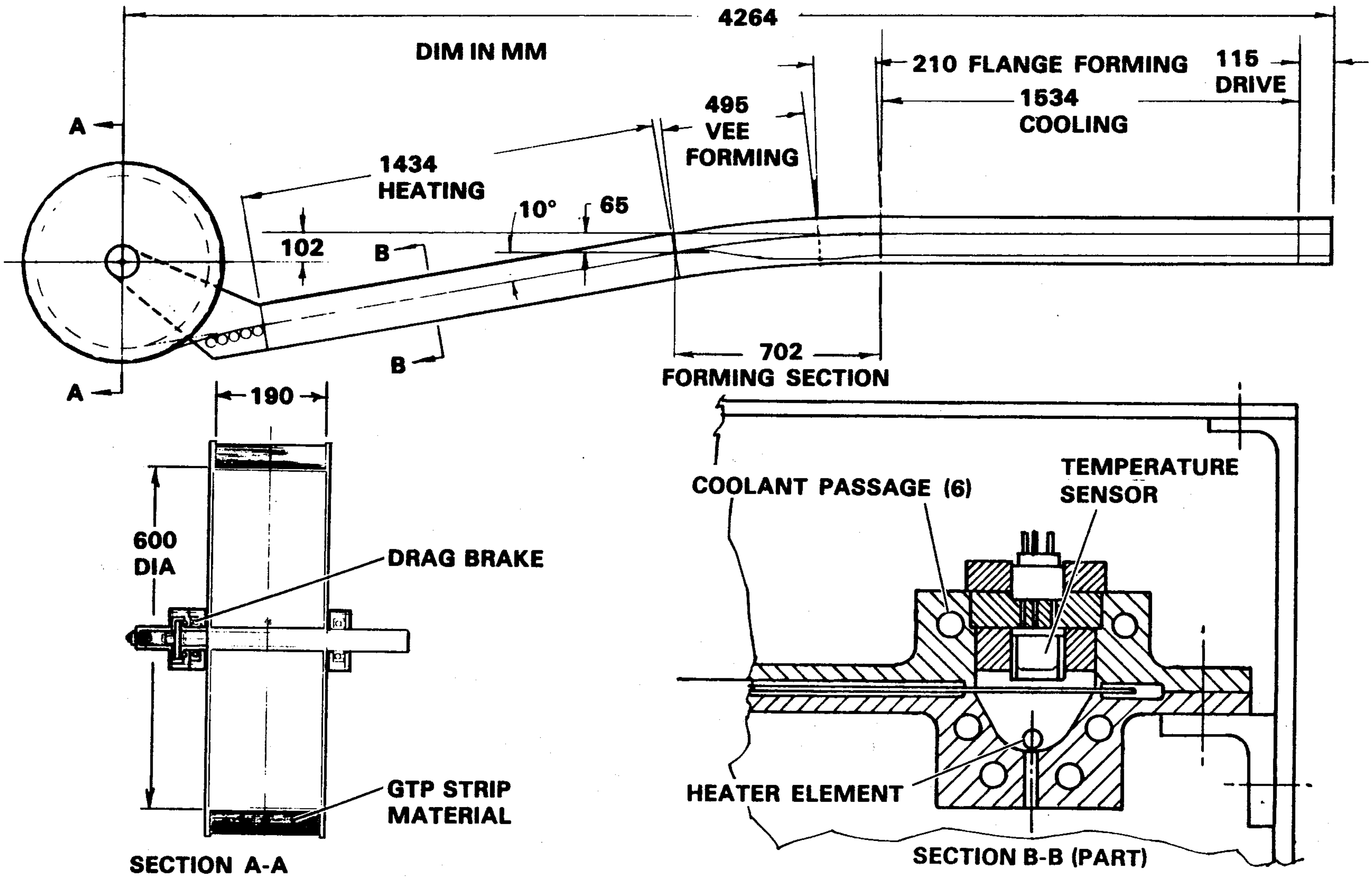


Figure 4

RADIANT HEATER PARAMETRICS AND DESIGN (Figure 5)

The recommended energy transfer mechanism is radiation from rod heaters. The radiation is directed at the fold areas to be heated, the other areas being masked. This method is used for primary operational heating, auxiliary start-up heating, and temperature maintenance. Parametric design data for the heater elements is given opposite.

Certain factors of uncertainty exist with regard to the emittance values used. A figure of 0.20 is available for non-oxidized nickel-chrome (80%, 20%) wire alloy and parameters in the region of 0.80 have been determined and reported for the alloy, when oxidized for 15 minutes at a temperature of 2100° F. The published emittance figures are closer to 0.86 at the temperature levels found herein. Emittance is significant with regard to definition of the stable operating temperature level of the nickel-chrome element wire.

Calculations showed that a wound heating element is feasible and applicable. What must yet be determined is the material which, both electrically and thermally non-conductive, can be formed or shaped to the established requirements. The chart illustrates these requirements, and also shows the relative arrangement/scale of the nickel-chrome wire lay. This basic design was found applicable to each of the strip heating applications.

RADIANT HEATER PARAMETRICS & DESIGN

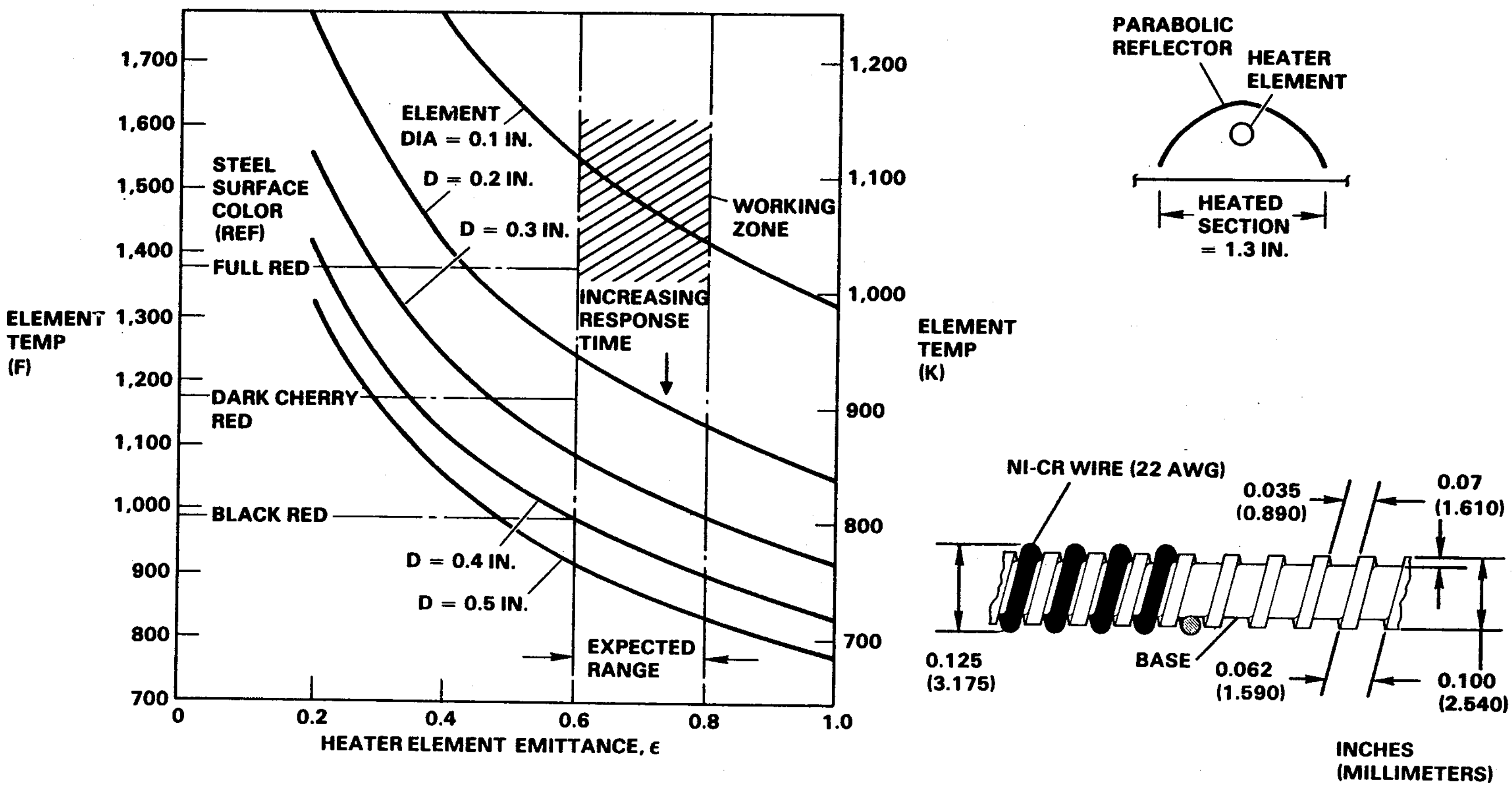


Figure 5

CAP SECTION FORMING (Figure 6)

The key feature of unidirectional graphite composite laminates is the inextensibility of the graphite fibers themselves. Even when the matrix is softened by heat, the fibers retain their original inextensible property; and in passing from Section A to Section F, shown in the chart opposite, they must maintain a conservatism of length. Experimental investigation shows that the free form between Section A and Section F of the material for a given length is explicit. By observation, the original shape-taking of the section shown in Section B is convex and there is a tension force applied at the softened apex of the section up to about 60% of the travel (i.e., Section C). The straight sections of the material pass through a point of inflexion, and, beyond there they become concave, giving rise to a compression at the softened apex. At Section B, the main apex angle of 60° is fully formed; and, between Sections E and F, the flanges may now be turned. Any attempt to turn the flanges at the same time as the apex angle is being formed will give rise to buckling of the softened material with a consequently greater probability of extensive delamination.

CAP SECTION FORMING

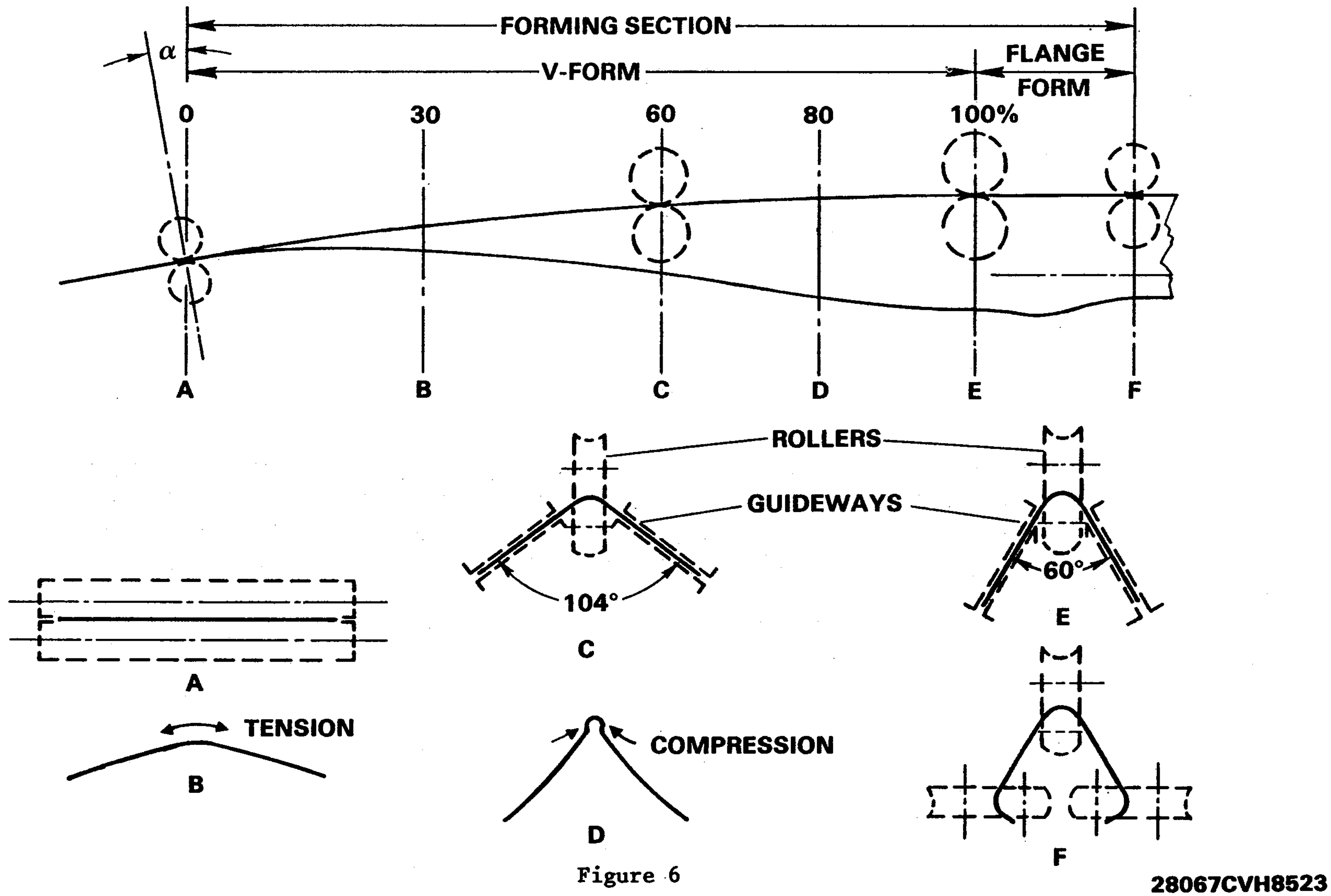


Figure 6

COOLING PARAMETRICS (Figure 7)

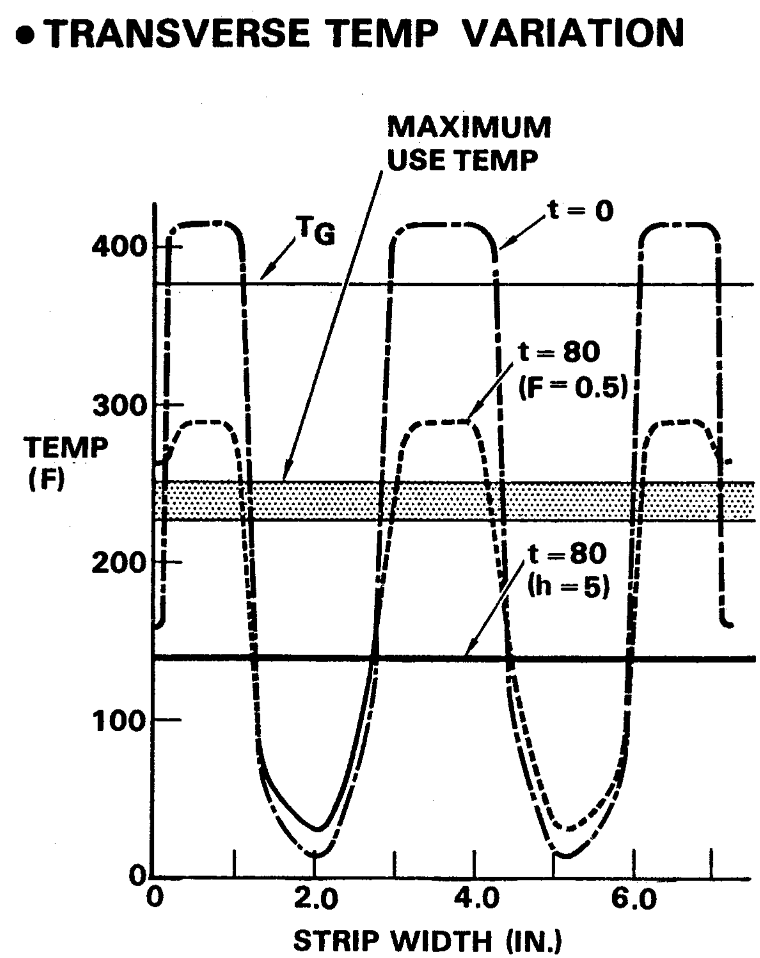
The characteristics of the temperature distribution in a hybrid laminate as it exits the beam builder forming section were investigated.

An analysis was conducted in which the strip was allowed to radiate (with a relatively optimistic deep space view factor, F, of 0.5) as well as conduct within itself over a complete 80-second machine cycle. As shown, the resulting peak temperatures still exceed the use temperature constraint, indicating that dedicated cooling is still required.

As seen in variations among key parameters controlling temperature decay rate, radiative/transverse conductive cooling is marginal even with a space view factor of 1.00.

Conversely, the platen cooling option permits significant reduction in pause cycle time (other processes permitting) even for a conservative contact coefficient of 5 $BTU/hr\text{-}ft^2\,°F$.

COOLING PARAMETRICS



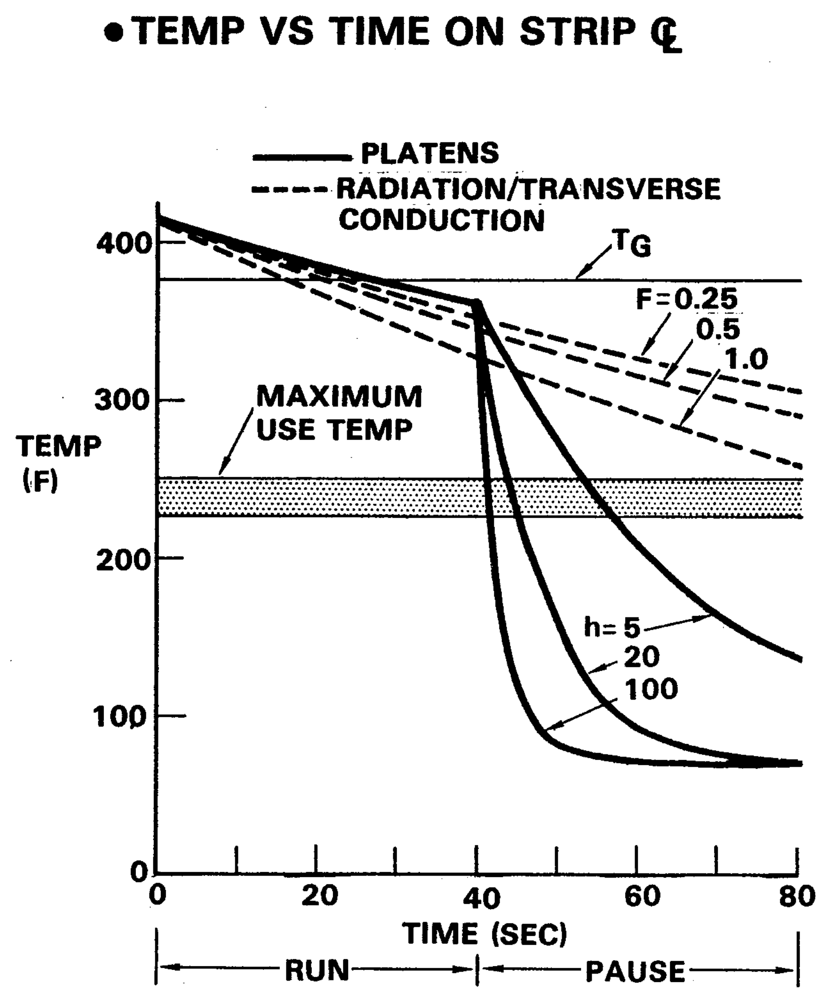


Figure 7

17018000H8729A

BEAM FABRICATOR GENERAL ARRANGEMENT (Figure 8)

This beam fabricator concept feeds and processes raw materials in the form of coiled graphite polysulfone laminate strip and polysulfone impregnated S-glass roving cord into a completed continuous braced triangular cross section beam of one meter depth. New technology in the rolltrusion hot forming of structural sections is combined with state-of-the-art mechanical and electrical/electronic design practices to effect straightforward flightweight product. Through an iterative design and trade study development approach, the machine shown in general arrangement opposite has been evolved.

Principal features of the beam fabrication machine are shown. Six reels of strip graphite/thermoplastic (GTP) material feed the rolltruder units which produce three continuous cap section members and the beam cross-members. The cross-members are cut to length positioned, and ultrasonically welded in place. Each bay of the beam is diagonally braced by crossing cords. These are dispensed and tensioned by an oscillating arm mechanism. The cords are fastened by welding them between the contact surfaces of cross-members and caps. Differential cord tensioning is used to correct any inherent twist in the beam. Straightness within construction tolerances is achieved by synchronized cap extension.

BEAM FABRICATOR GENERAL ARRANGEMENT

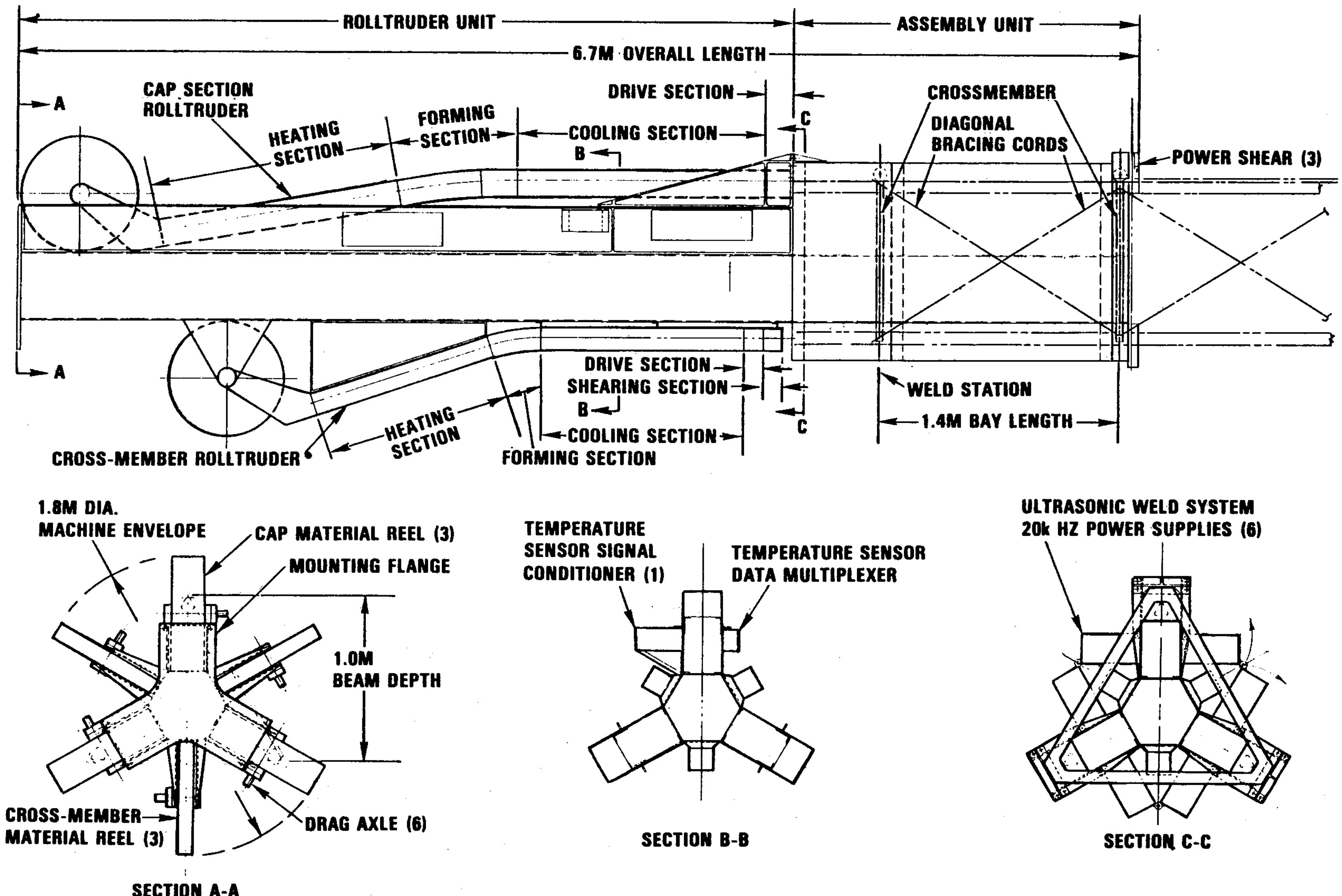


Figure 8

ULTRASONIC ASSEMBLY WELDING (Figure 9)

In a spot weld, the horn normally used contains a "piercing tip" or pilot which penetrates through the top sheet. The molten plastic displaced on the spot weld surface is shaped by a radial cavity in the tip and forms a raised ring of thermoplastic material simultaneously, energy is released at the interface, producing essential frictional heat to cause a thermoplastic melting and flowing to form a permanent molecular bond.

The chart shows a cross section of a flat weld and a typical piercing spot weld.

The ultrasonic spot weld method is the most attractive joining system investigated for our beam-building system. It is rapid, uses a minimum amount of energy, can be designed in an acceptable size and weight package, and produces excellent welds using a variety of weld tips. The type of tips and supporting mechanical and electrical combinations are numerous and varied. The chart illustrates several possible weld configurations. The selection of any weld configuration should be based on design requirements and energy limitations.

ULTRASONIC ASSEMBLY WELDING

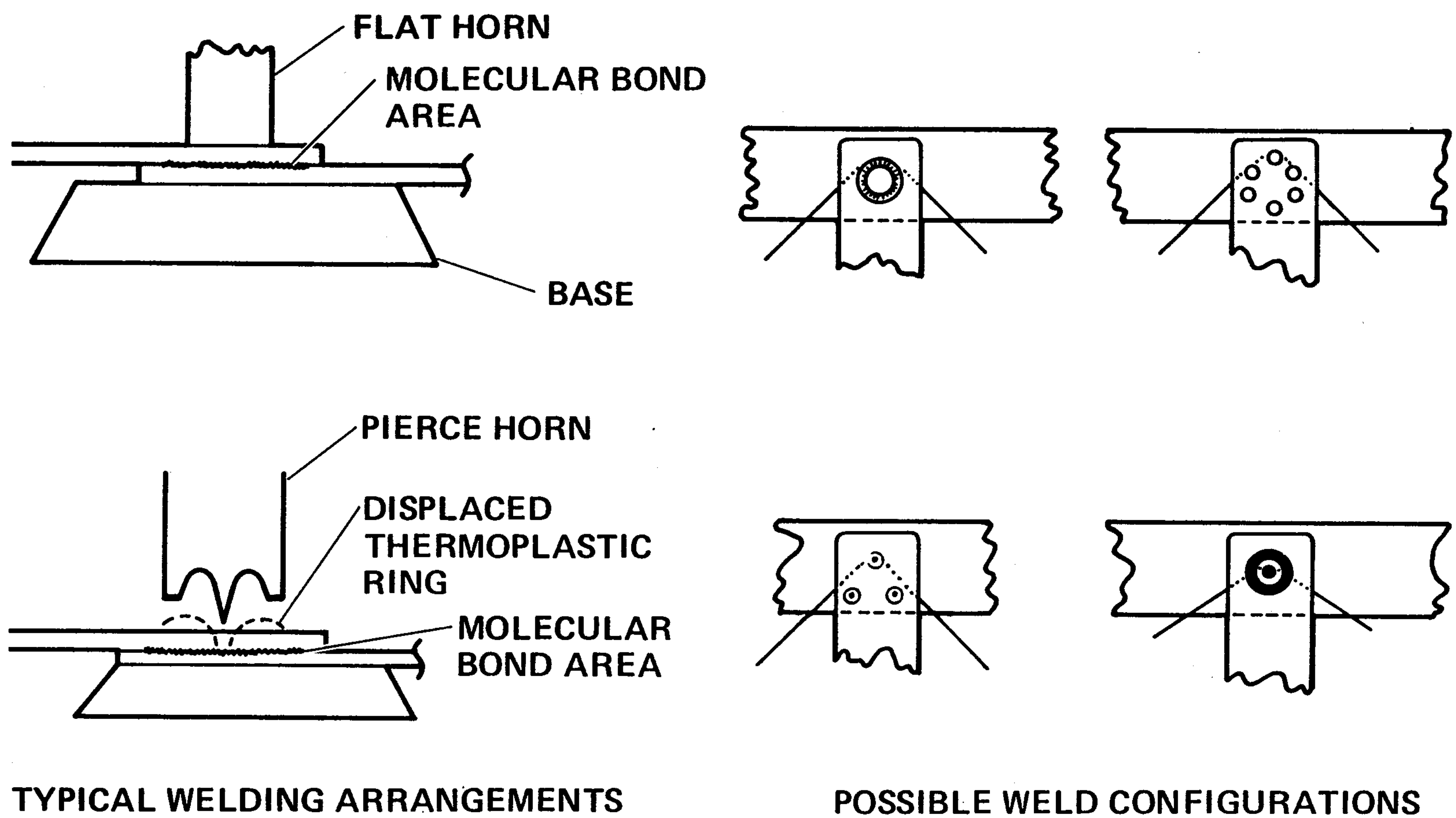


Figure 9

BEAM FABRICATION MACHINE - ASSEMBLY SECTION (Figure 10)

This perspective view of the beam fabricator assembly section shows the major functioning mechanisms. The three cap sections emerge from their rolltrusion units. Cross members have been sheared at the exit from their rolltruders and are partially swung into position for welding. The dispensing arms have laid cord diagonals in position for cross member placement beneath the articulated ultrasonic weld heads. At the machine exit, locations of the beam cap shears are apparent. Shearing anvils are firmly supported from the hexagonal section structural spine.

BEAM FABRICATION MACHINE — ASSEMBLY SECTION

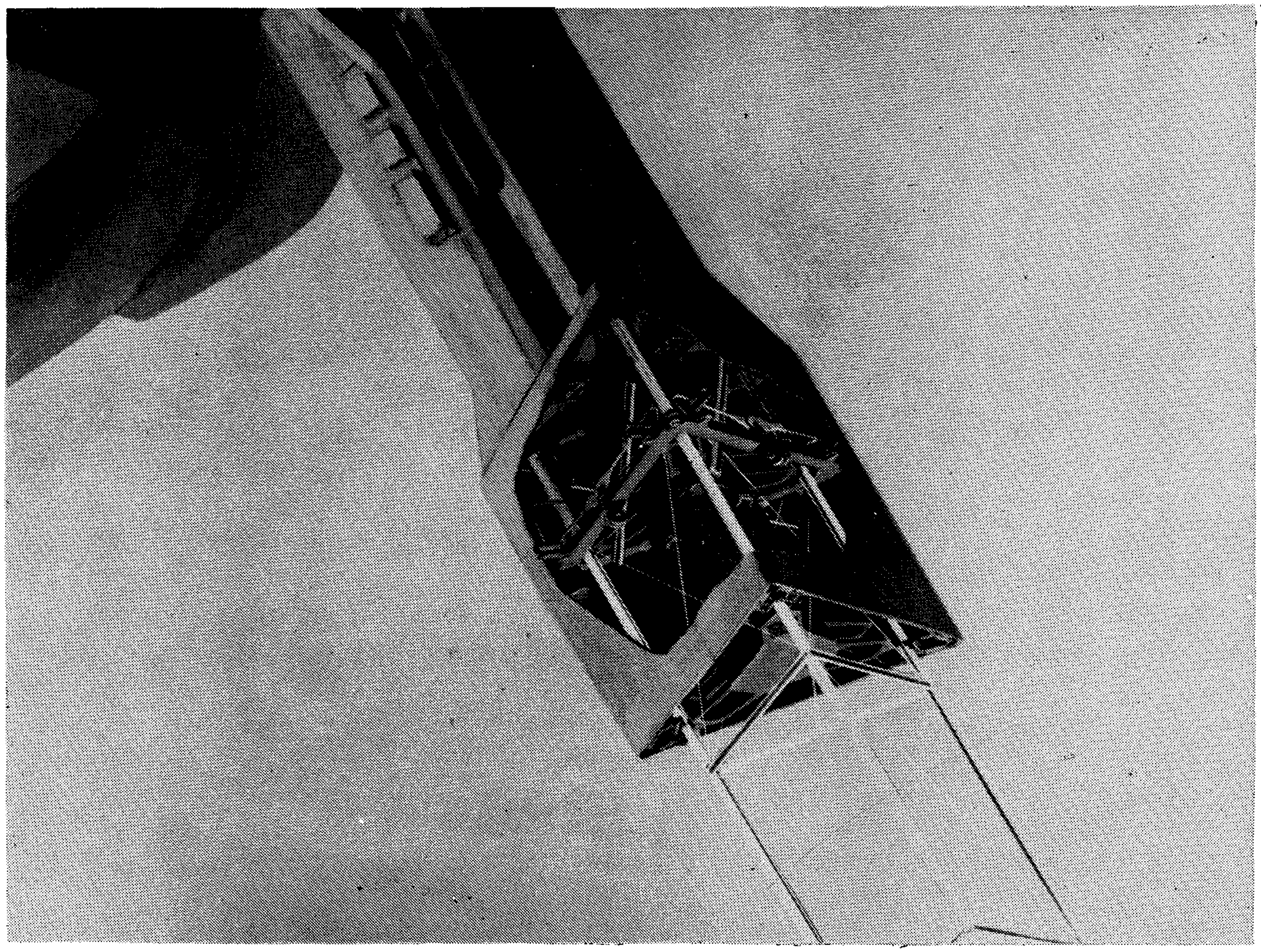


Figure 10

07127CVH8979

TYPICAL JIG & BEAM BUILDER DEPLOYMENT SEQUENCE (Figure 11)

The assembly jig is deployed by unlatching the forward Z support pins and rotating the jig about an axis concentric with the aft X-Z trunion support pin.

When the longitudinal axis of the jig is parallel to the Z axis, the jig is locked into position and the beam builder is unlatched for deployment.

Beam builder deployment and positioning is performed as a series of operations by the roll and turn mechanism. The beam builder is rolled 180° in two steps as shown in Steps 3 and 4. It is then turned 90° to the orientation required for longitudinal beam fabrication as shown in Step 5.

To reorient the beam builder for cross-beam fabrication, it is first turned back 90°. The roll link then rotates 180° as the beam builder counter rotates 120° resulting in a net rotation of the beam builder of 60° and a lateral translation to the desired position.

TYPICAL JIG & BEAM BUILDER DEPLOYMENT SEQUENCE

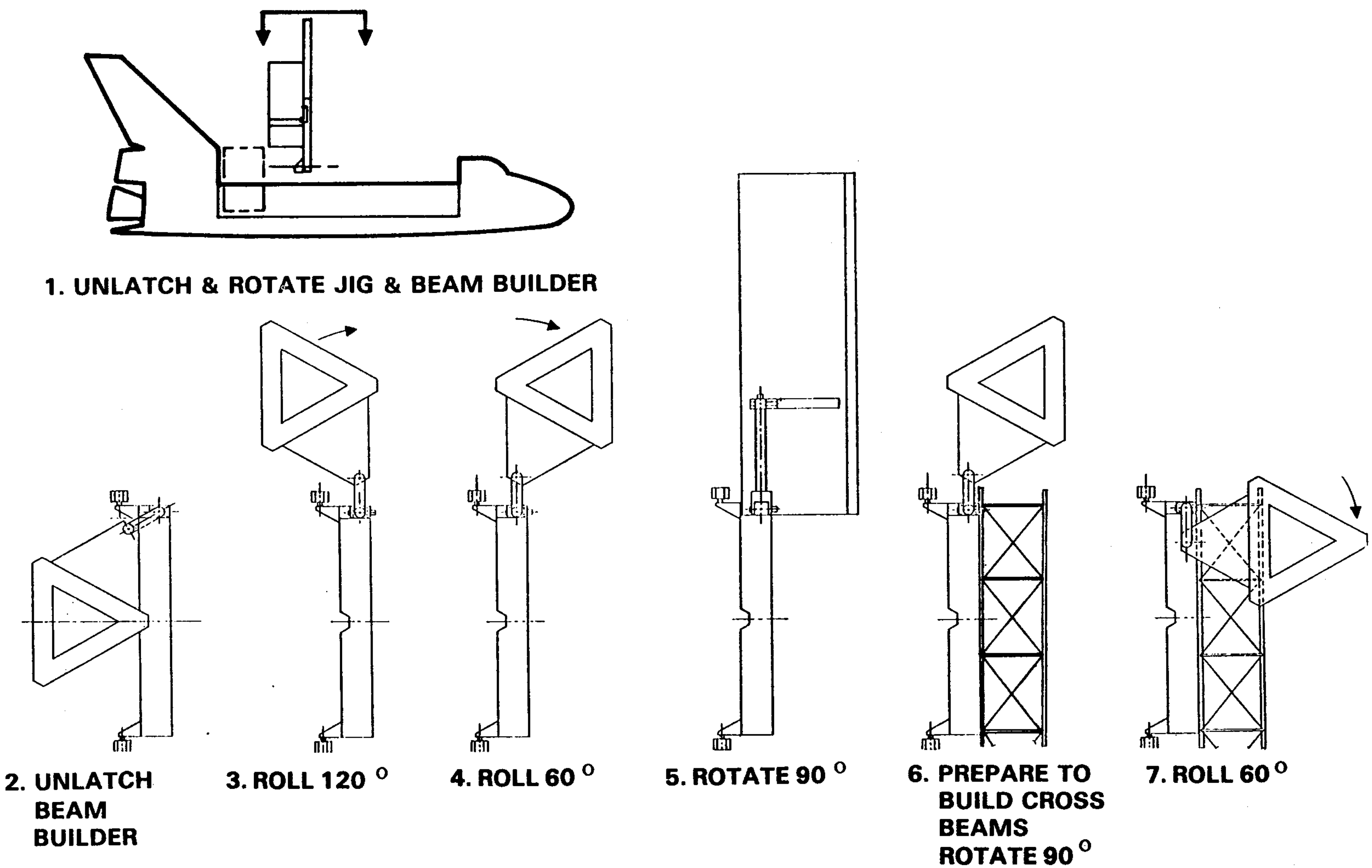


Figure 11

CROSS-BEAM TO LONGITUDINAL BEAM JOINING TECHNIQUES (Figure 12)

Use of EVA personnel for performing cross-beam to longitudinal beam joining is considered too time consuming and would interfere with other EVA activities where manual action is more advantageous; e.g., installing instrumentation or connecting instrument wires. Highly repetitive operations such as beam joining should be automated wherever practicable.

Automatic welding options considered are compared as follows:

<u>Single Row Traveling Welder</u>

- o Greatest control complexity
- o Slowest joining method (mission impact)
- o Adds a special carriage mechanism
- o Minimum number of weld heads for least power utilization

<u>Double Row Fixed Welder</u>

- o Fastest joining method
- o Maximum number of weld heads (16) increases power requirements
- o Simplest welding mechanism
- o Least number of control functions

<u>Single Row Fixed Welder</u>

- o Permits welding operations within reasonable time limits with fewer weld heads (8)
- o Half the power usage of double row welders
- o Adds index drive mechanism to welding mechanism
- o Adds to complexity of platform drive control

CROSS BEAM TO LONGITUDINAL BEAM JOINING TECHNIQUES

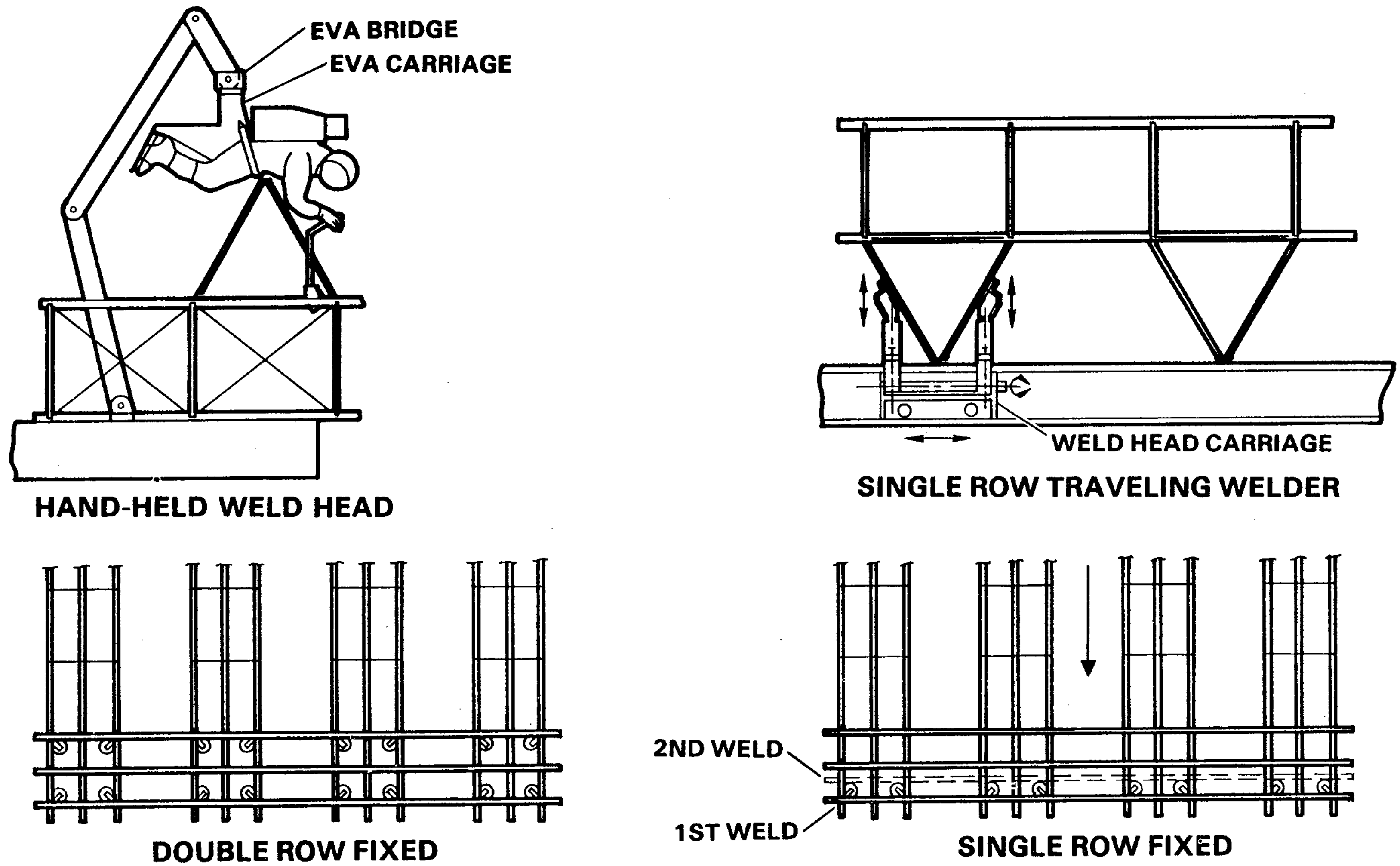


Figure 12

MSFC PRESENTATION CHARTS

ON

GEOSTATIONARY PLATFORM

PRESENTED AT THE

GOVERNMENT/INDUSTRY SEMINAR ON LARGE SPACE SYSTEMS TECHNOLOGY

BY

WILLIAM T. CAREY, JR.

JANUARY 1978

GEOSTATIONARY PLATFORM (Figure 1)

(BACKGROUND)

The Geostationary Platform has been shown to be a more cost effective way of accomplishing a wide variety of geosynchronous missions in lieu of the traditional specialized satellite approach. The platform (which will be assembled in low earth orbit from elements supplied by multiple Shuttle launches for subsequent transfer to geostationary orbit), could fly as soon as 1986.

The idea of a Geostationary Platform is not new for it has been the topic of speculation and study for several years. However, now that people throughout NASA, the Aerospace Community and the potential user community have seen the Shuttle in flight, they are beginning to realize that the capability to build such large structures in space is upon us.

The concept as presented herein has been built on a set of missions provided to us by the Office of Space and Terrestrial Applications. We have also worked with them in developing the concept. Many of you have seen some of the Advanced Concepts developed by various NASA groups and their contractors. They are frequently called Public Service Platforms. Most of these concepts are extremely large and special purpose, and thus are candidates for the decade of the 90's. The platform I am talking about would be much earlier and would be multipurpose. In many ways, the Geostationary Platform can be viewed as a precursor to these large special purpose satellites.

We believe that in configuring a Geostationary Platform institutional considerations are very important. As you will see, we have maintained a clean interface between what the users would provide and the platform. Some of the benefits that a platform could offer in comparison to conducting the same missions via the traditional specialized satellite mode are shown on Figure 1.

ORGANIZATION: PROGRAM DEVELOPMENT	MARSHALL SPACE FLIGHT CENTER GEOSTATIONARY PLATFORM (BACKGROUND)	NAME: CAREY DATE: JANUARY 1978

- IDEA OF A GEOSTATIONARY PLATFORM IS NOT NEW: HOWEVER, THE UTILITY AND POTENTIAL OF THE STS (INCLUDING LARGE SPACE STRUCTURES, MAN-IN-SPACE AND GEOSTATIONARY CAPABILITIES) SEEMS TO BE MORE RECOGNIZED NOW.

- WORKING WITH THE OFFICE OF SPACE AND TERRESTRIAL APPLICATIONS (OSTA) IN DEFINING CONCEPT.

- CONCEPT WHICH SEEMS TO MAKE THE MOST SENSE FOR EARLY (1986-1988) APPLICATION TAKES FORM OF A STRONGBACK STRUCTURE WITH ANTENNAS AND OTHER PAYLOADS MOUNTED THEREON.

  - VERY LARGE ANTENNAS TO FOLLOW LATER
  - INITIAL PLATFORM OVER CONUS - OTHERS FOLLOW LATER

- INSTITUTIONAL CONSIDERATIONS ARE VERY IMPORTANT

- SOME OF THE BENEFITS ARE:

  - ECONOMIES THAT CAN BE PASSED ON TO USERS
  - ON-BOARD SWITCHING BETWEEN ANTENNAS-FLEXIBILITY
  - LARGER APERTURE ANTENNAS PERMITTING LOWER COST GROUND TERMINALS
  - MORE EFFICIENT USE OF ALREADY OVERCROWDED SPECTRUM
  - OPPORTUNITY FOR NEW MISSIONS NOT OTHERWISE AVAILABLE

Figure 1

GEOSTATIONARY PLATFORM (Figure 2)
(POTENTIAL LONGITUDINAL LOCATIONS)

Figure 2 shows the longitudinal location of the operational geostationary satellites currently in orbit. The satellites tend to group in four longitudinal positions: the United States, Western Europe, India and the Pacific. Furthermore, within each of these positions, the satellites generally prefer the central longitudinal position of the geophysical area they serve so that they can communicate with any ground station with elevation angles of greater than 5°.

In order to avoid interference, satellites must have a separation distance of at least 4° if they operate in the same frequency band. We are simply beginning to run out of parking spaces. As one looks to the decade of the 1980's, the problem will become much worse. The Geostationary Platform with its inherent capability to provide large aperture antennas with multiple, narrow beams and the ability to switch signals from beam-to-beam and antenna-to-antenna provides an answer to this dilemma. Furthermore, we believe that by taking the Geostationary Platform approach instead of the traditional individual satellite approach, cost savings can be realized and passed on to users.

The missions and the concept that I will be showing today are for the platform over the United States. Platforms over the other high density traffic areas would probably look much the same.

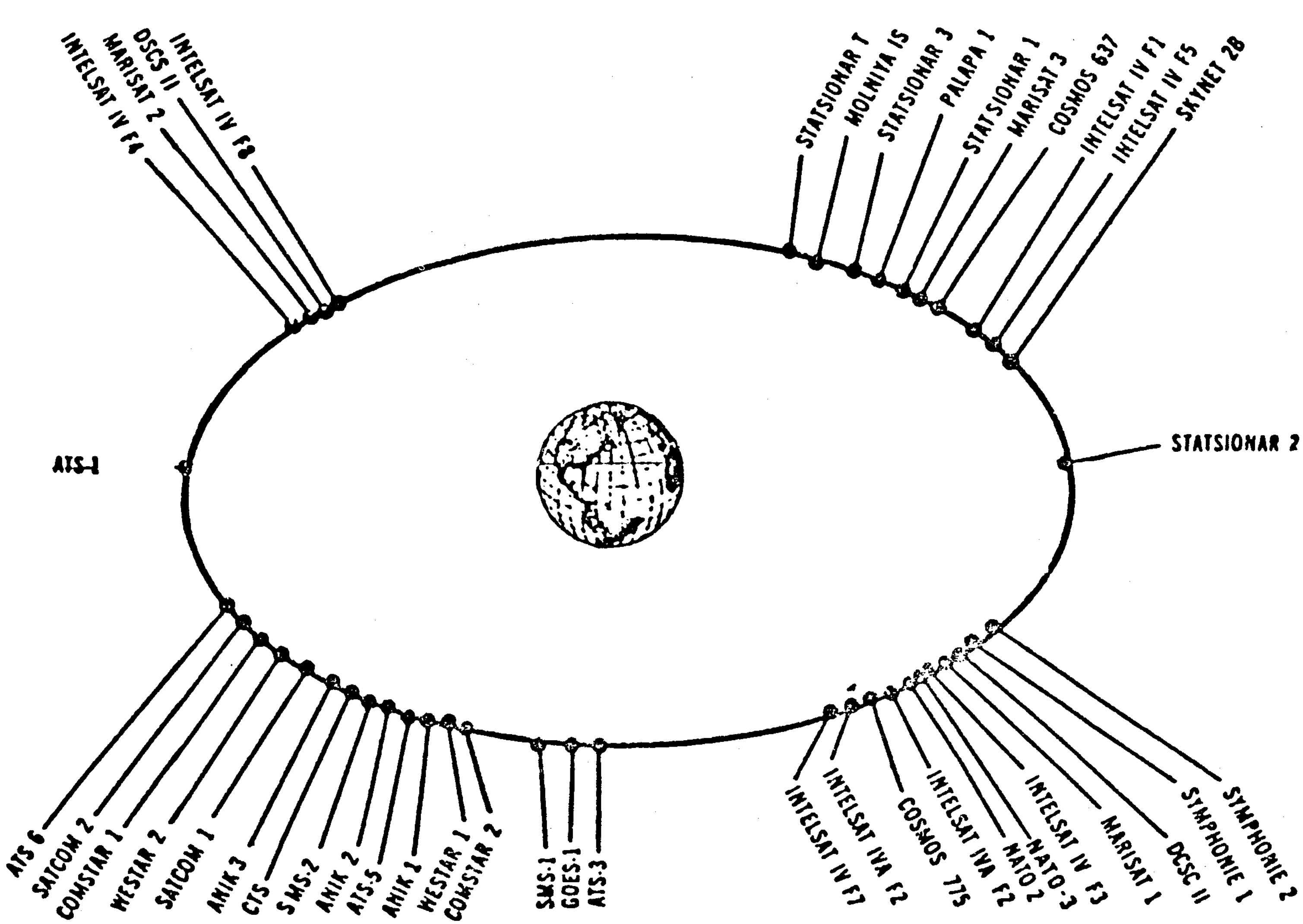

INTELSAT IV F8
DSCS II
MARISAT 2
INTELSAT IV F4
STATSIONAR T
MOLNIYA 1S
STATSIONAR 3
PALAPA 1
STATSIONAR 1
MARISAT 3
COSMOS 637
INTELSAT IV F1
INTELSAT IV F5
SKYNET 2B
STATSIONAR 2
ATS-1
ATS 6
SATCOM 2
COMSTAR 1
WESTAR 2
SATCOM 1
ANIK 3
CTS
SMS-2
ANIK 2
ATS-5
ANIK 1
WESTAR 1
COMSTAR 2
SMS-1
GOES-1
ATS-3
INTELSAT IV F7
INTELSAT IVA F2
COSMOS 775
INTELSAT IVA F2
NATO 2
NATO-3
INTELSAT IV F3
MARISAT 1
DCSC II
SYMPHONIE 1
SYMPHONIE 2

FIGURE 2

GEOSTATIONARY PLATFORM (Figure 3)
(DESIRED MISSION CAPABILITIES)

The next several charts show the candidate missions, which were identified by the Office of Space and Terrestrial Applications (OSTA). which we have used to derive the design of the Geostationary Platform. The four mission categories are shown on Figure 3.

GEOSTATIONARY PLATFORM
DESIRED MISSION CAPABILITIES

- FIXED POINT (POINT-TO-POINT) COMMUNICATIONS
- MOBILE SATELLITE COMMUNICATIONS SERVICE
- BROADCAST SATELLITE SERVICES
- SPACE RESEARCH, METEOROLOGY AND EARTH OBSERVATION SATELLITES

FIGURE 3

GEOSTATIONARY PLATFORM DESIRED MISSION CAPABILITIES (Figure 4)

FIXED (POINT-TO-POINT) COMMUNICATIONS

Missions in the area of fixed (point-to-point) communications which could be accommodated on the platform are shown on Figure 4. The two C-Band services shown at the top of the page would accomplish the next generation function now being provided by the current domestic communications satellites (COMSTAR, WESTAR, SATCOM), the Canadian ANIK, and ATS-6. New services and capabilities are also provided at S-Band, Ku-Band, and K-Band.

GEOSTATIONARY PLATFORM
DESIRED MISSION CAPABILITIES

FIXED POINT (POINT–TO–POINT) COMMUNICATIONS

TWO C–BAND BROAD BEAM ANTENNAS FORMED BY A 0.7M X 2.0M ELLIPSOIDAL REFLECTOR COVERING ALL CONUS TO ALLOW ANY EARTH STATION TO COMMUNICATE WITH ANY OTHER EARTH STATION (4.5M DIAMETER TRUNKLINE GROUND RECEIVERS).

ONE 30M C–BAND ANTENNA WITH MULTIPLE FEEDS PROVIDING 37 SPOT BEAMS TO CONUS METRO AREAS AND 3 SPOT BEAMS TO HAWAII, ALASKA, AND PUERTO RICO. (4.5M DIAMETER TRUNK LINE RECEIVERS).

THESE SERVICES CURRENTLY PROVIDED VIA 10 INDIVIDUAL DOMESTIC SATELLITES IN GEOSTATIONARY ORBIT WHICH MUST MAINTAIN $\gtrsim 4^{\circ}$ SEPARATION TO AVOID MUTUAL JAMMING.

ONE 12M DIAMETER KU–BAND ANTENNA WITH MULTIPLE FEEDS PROVIDING BROADCAST TO THE 20 LARGEST METRO AREAS ANYWHERE WITHIN CONUS (1.5M ROOFTOP RECEIVER ANTENNAS).

FOUR INDEPENDENTLY GIMBALLED 4.5M DIAMETER Ku–BAND ANTENNAS COVERING FOUR SEPARATE GEOGRAPHIC AREAS WITH NEWS OR SPORTING EVENTS.

ONE 1M X 3M S–BAND ANTENNA TO PROVIDE CONUS COVERAGE FOR "THIN ROUTE" POINT–TO–POINT COMMUNICATIONS. (3M DIAMETER GROUND RE–CEIVER).

SMALL HORN ANTENNA (CONUS COVERAGE) AND A 2M DIAMETER ANTENNA WITH MULTIPLE FEEDS (10 BEAMS) TO PROVIDE K–BAND TO MAJOR METROPOLITAN AREAS TO SUPPORT EXPERIMENTS IN ELECTRONIC MAIL, DATA TRANSFER AND PERSONAL COMMUNICATIONS.

ELECTROMAGNETIC SPECTRUM (REF)

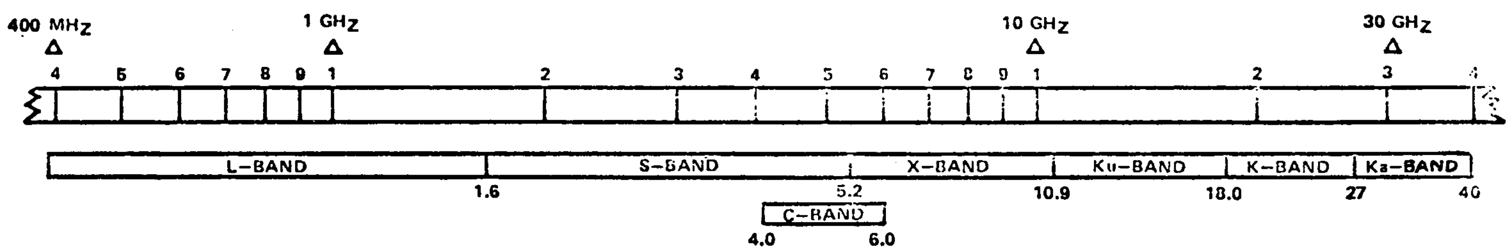


Figure 4

GEOSTATIONARY PLATFORM
DESIRED MISSION CAPABILITIES (Figure 5)
(MOBILE SATELLITE COMMUNICATIONS SERVICE)

The mobile satellite communications services are to be accommodated on the platform are shown on Figure 5.

GEOSTATIONARY PLATFORM
DESIRED MISSION CAPABILITIES

(MOBILE SATELLITE COMMUNICATIONS SERVICE)

FOUR 3M DIAMETER L–BAND ANTENNAS PROVIDING DIRECT LINK BETWEEN SHIPS AND LAND BASED TERMINAL (IMPROVED SERVICE COMPARED TO CURRENT MARISAT AND PLANNED MAROTS).

ONE 10M (UHF) ANTENNA AND ONE 1M (L–BAND) ANTENNA PROVIDING COMMUNICATIONS BETWEEN AIRCRAFT AND LAND BASED TERMINAL AS PREVIOUSLY CONSIDERED FOR AEROSAT.

ONE 12M DIAMETER L–BAND ANTENNA WITH 4 FEEDS PROVIDING CONUS COVERAGE FOR COMMUNICATIONS BETWEEN TRUCKS, CARS, ETC.

ELECTROMAGNETIC SPECTRUM (REF)

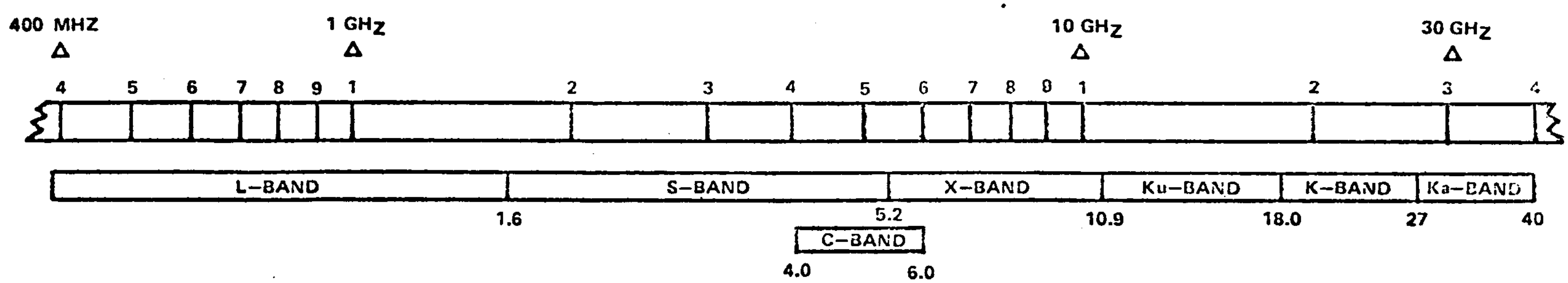


Figure 5

GEOSTATIONARY PLATFORM (Figure 6)
(TYPICAL EXAMPLE OF IMPROVED SERVICE)

The MARISAT, which is currently in operation; provides communication links between ships and land terminals. The current system, and that planned for the next generation (MAROTS) requires that the link pass through both the satellite and a large earth station. The switching function to assure the right ship is connected to the right land terminal is done in the earth station.

In the Geostationary Satellite Mode the switching function would be done in the satellite, and thereby eliminates the need for the earth station. The land terminal would link directly with the Platform through the appropriate narrow beam antenna. On board the platform the signal would be routed to the proper antenna and beam which covers the region in which the ship is located.

The resulting cost savings made possible by eliminating earth stations would enable the user to provide the service at a much lower cost.

This example is but one of many that a Geostationary Platform with on-board-switching could make possible.

GEOSTATIONARY PLATFORM
TYPICAL EXAMPLE OF IMPROVED SERVICE

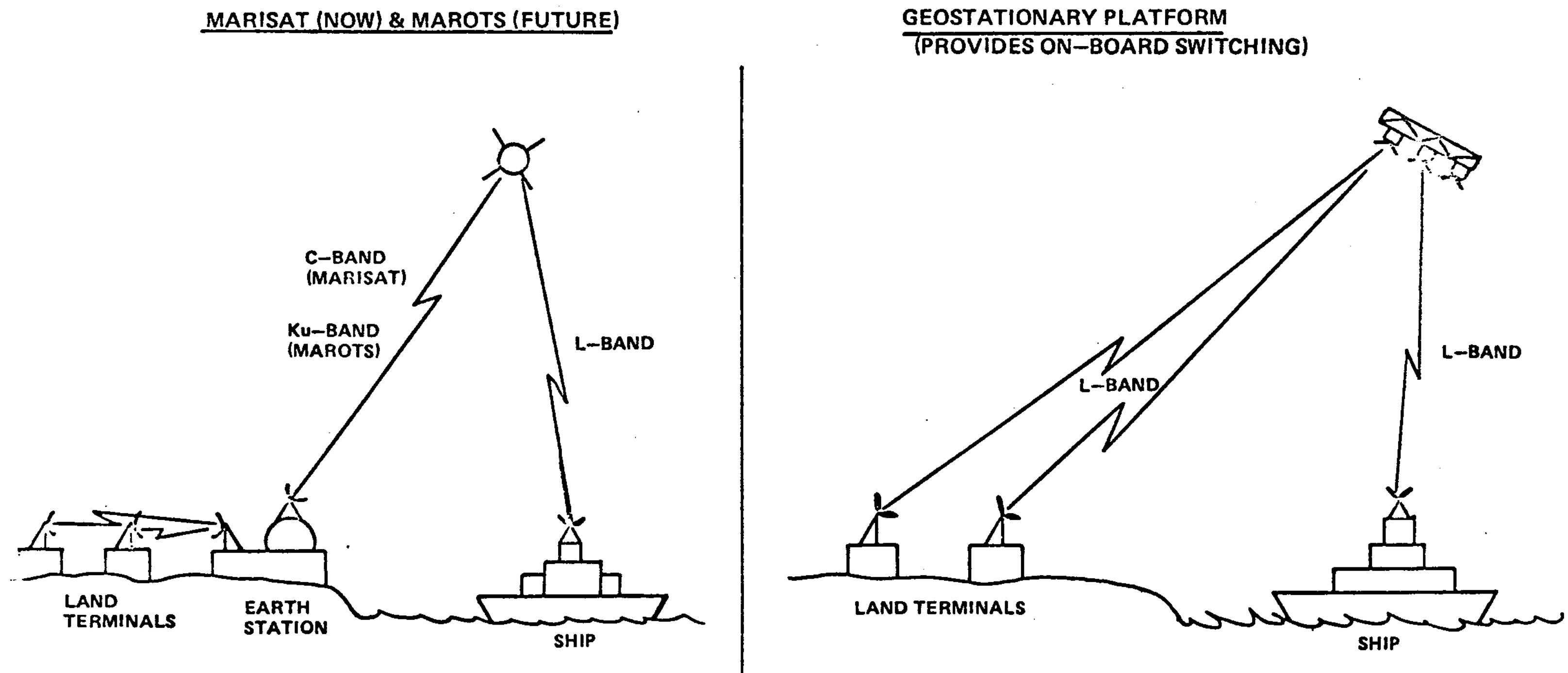


Figure 6

GEOSTATIONARY PLATFORM
LAUNCH AND ASSEMBLY SEQUENCE
(Figure 7, Figure 8, and Figure 9)

The Platform would be assembled in low earth orbit from elements supplied by multiple Shuttle launches. The first launch (Figure 7) carries the 30 meter parabolic antenna and other supplies to low altitude earth orbit. During the seven day mission, the 30-meter parabolic antenna is assembled. The second (Figure 8) launch carries the remaining parts to build-up the platform. The platform is assembled and checked out during the second flight. The third launch (Figure 9) provides the Geosynchronous Transfer Stage. After assembly of the basic platform and support systems and the attachment of mission payloads, the platform is transferred to geosynchronous orbit to a longitudinal position to serve missions over the United States. The platform would have an open ended lifetime with periodic visits by either manned or automated vehicles to carry out periodic or emergency repair and refurbishment and to bring on board new mission payloads.

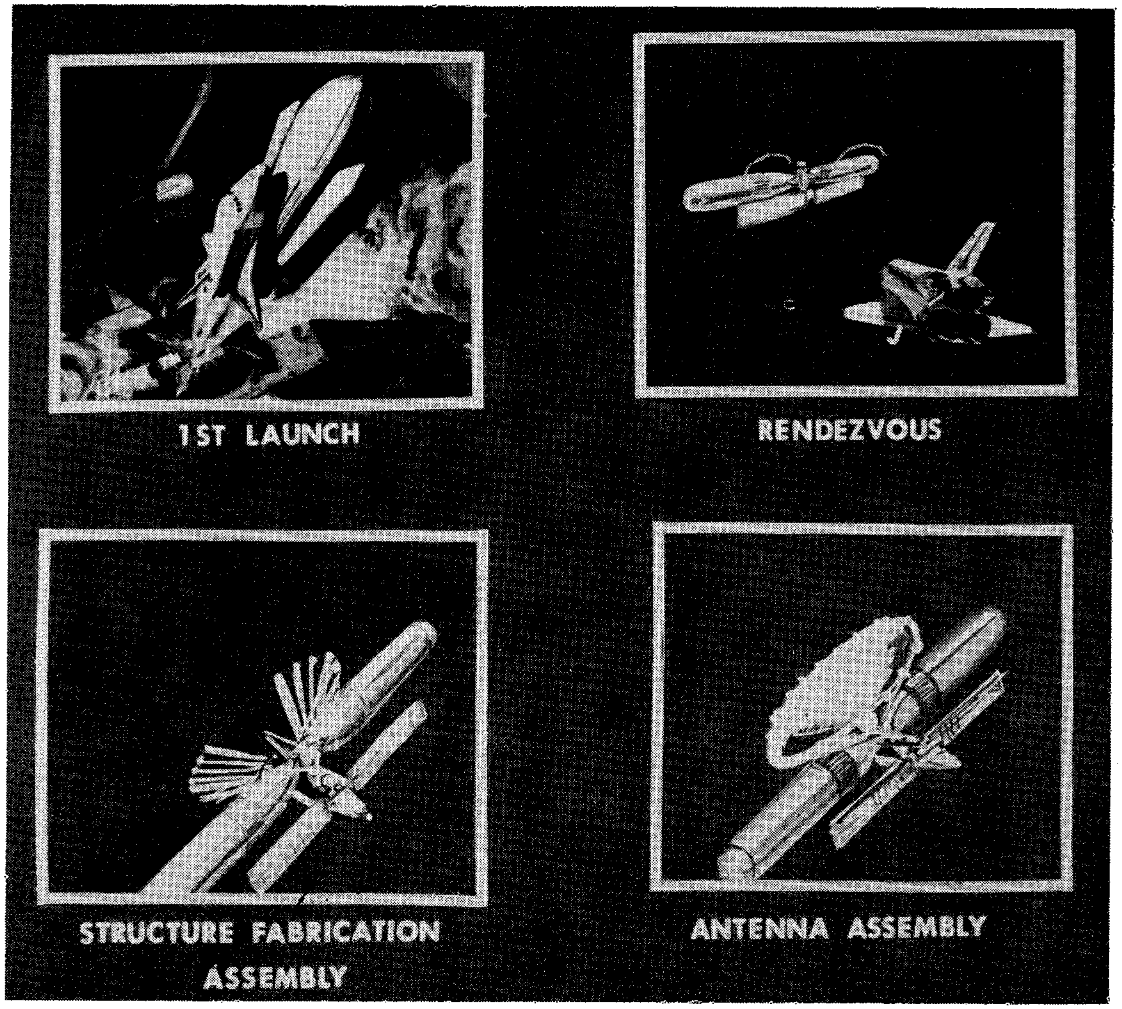

1ST LAUNCH
RENDEZVOUS
STRUCTURE FABRICATION ASSEMBLY
ANTENNA ASSEMBLY

Figure 7

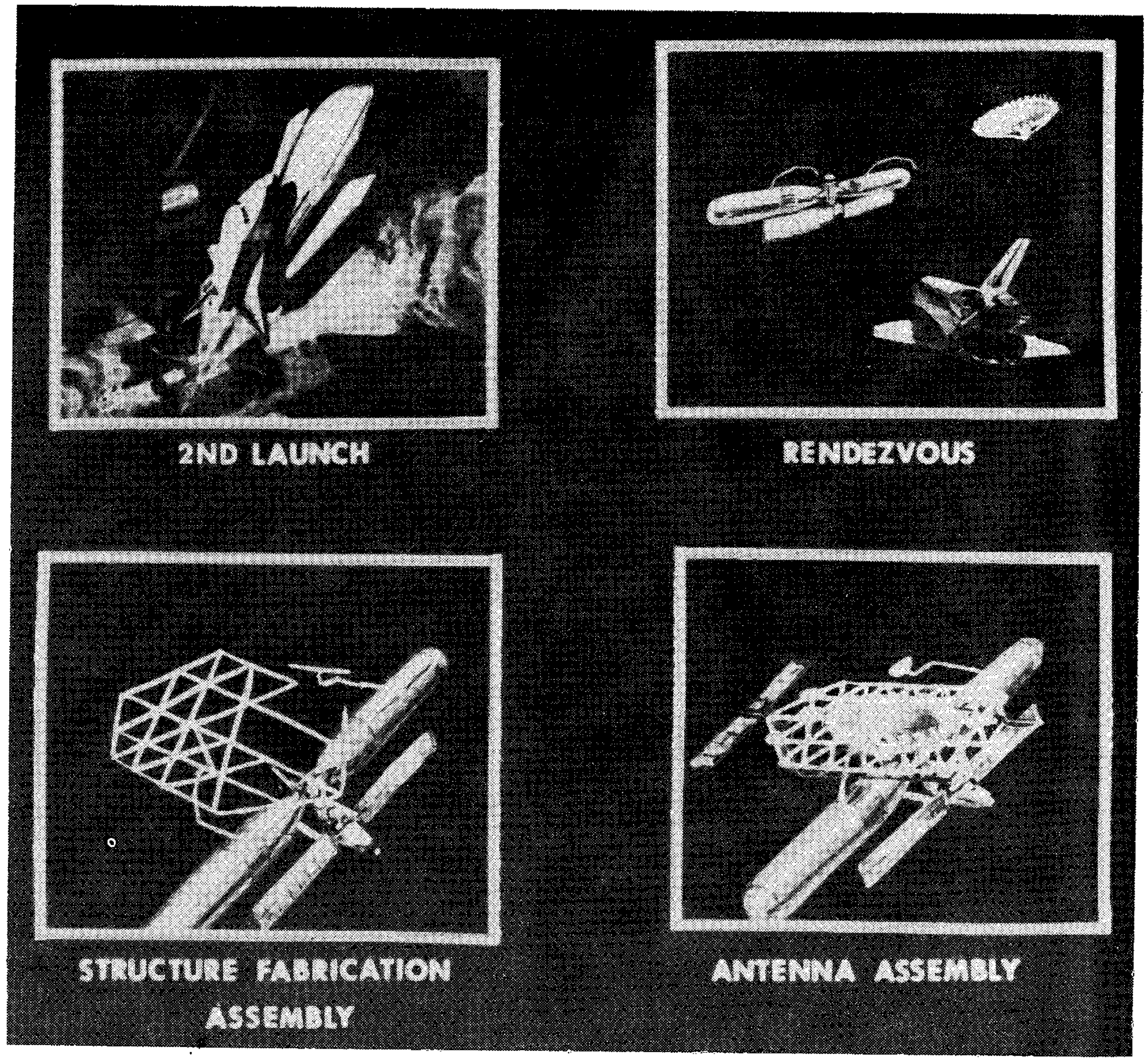

2ND LAUNCH
RENDEZVOUS
STRUCTURE FABRICATION ASSEMBLY
ANTENNA ASSEMBLY

Figure 8

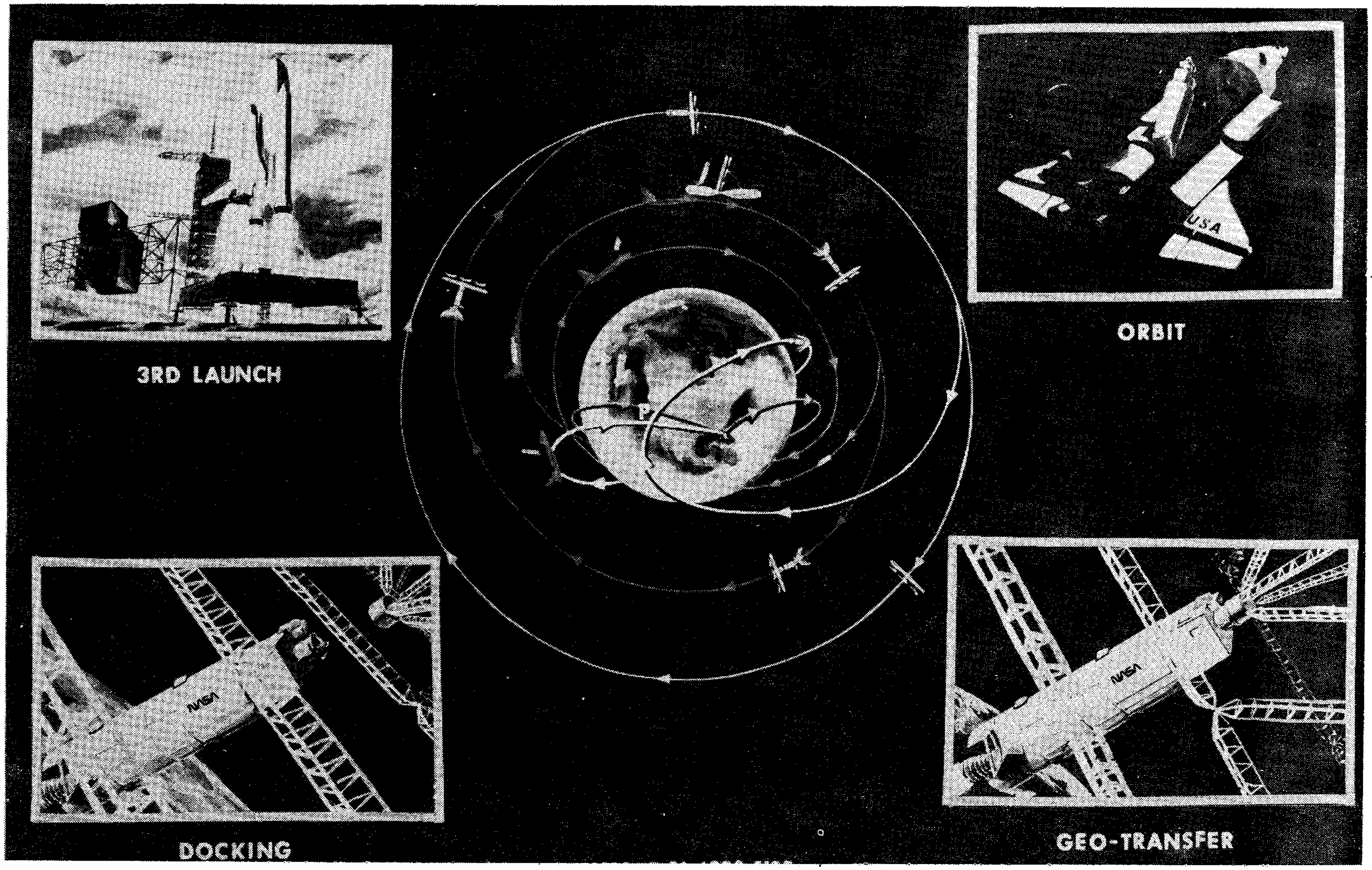


Figure 9

GEOSTATIONARY PLATFORM (Figure 10) (CONCEPT)

The Geostationary Platform as it would look while operating at its geostationary position is shown on Figure 10. The dimension of the platform are 82 meters by 31 meters and it will weigh around 18,000 pounds. Power modules with roll up solar arrays are shown at each end. A support module would be located on the back of the platform immediately behind the 30-meter antenna. This module would house switching equipment required to interconnect each antenna to the other and to other platform support equipment. Attitude control is maintained using 4 reaction wheels with periodic momentum dumping provided by the propulsion modules shown at 4 locations on the platform. Platform pointing would be provided at $\pm 0.5^{\circ}$ with fine pointing provided by each payload as required. Interface between the platform and the user-provided payloads has been kept simple with only structural attach points and electrical interface for power transfer and payload housekeeping functions. Thermal control is provided by each payload.

Most of the mission packages would be provided by the users who would pay a transportation fee for delivery to the platform and a "use" fee for drawing support (electrical power, coarse pointing, housekeeping functions, etc.) from the platform.

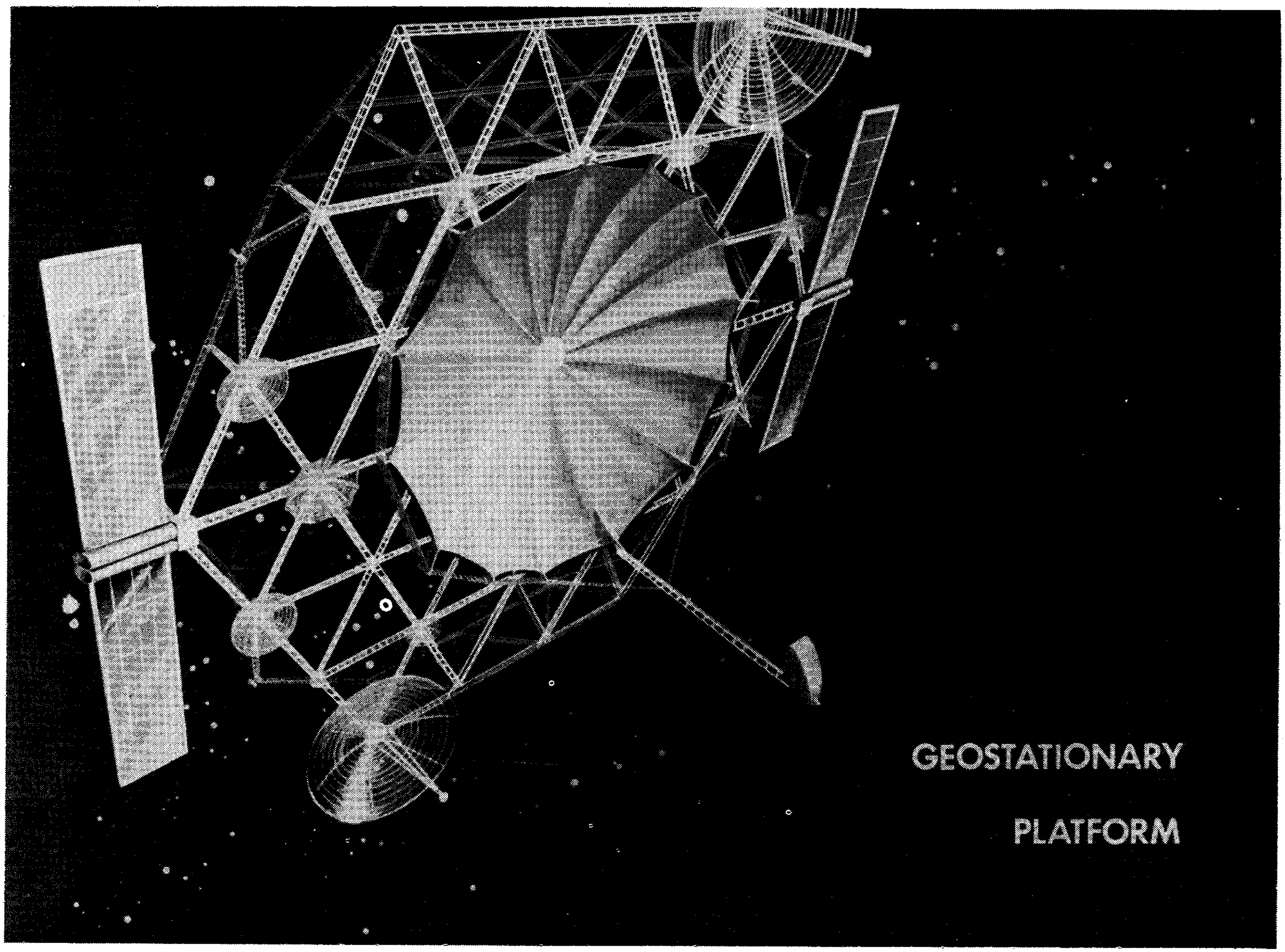


Figure 10

GEOSTATIONARY PLATFORM (Figure 11)

30-METER ANTENNA

One of the key features of the platform is a 30-meter parabolic antenna which would operate at C-Band and provide 40 spot beams to earth, each of which would be approximately 70 miles diameter. These beams would be made possible by providing an offset feed with 40 feed horns as depicted in Figure 11. These 40 spot beams would provide coverage to the major metropolitan areas of the United States. The preferred assembly or deployment approach for this antenna is currently under study.

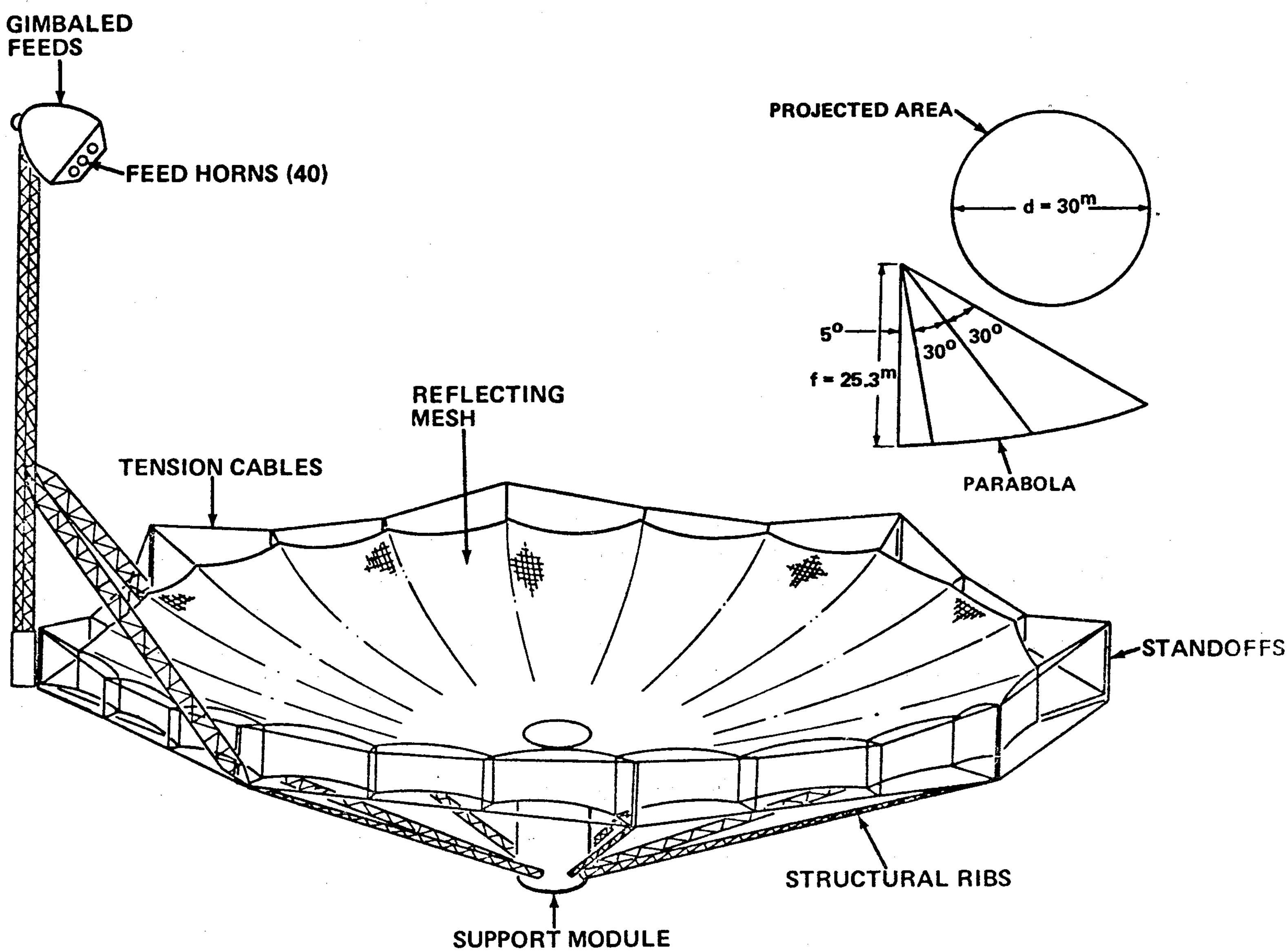

GIMBALED FEEDS
FEED HORNS (40)
PROJECTED AREA
d = 30m
5°
30°
30°
f = 25.3m
PARABOLA
REFLECTING MESH
TENSION CABLES
STANDOFFS
STRUCTURAL RIBS
SUPPORT MODULE

FIGURE 11

GEOSTATIONARY PLATFORM (Figure 12)

(CONCLUSIONS)

In summary, we believe the Geostationary Platform has great promise. Although we are now only in the preliminary definition phase we believe that we have penetrated to sufficient depth to say that there are no showstoppers. Our plans are to continue our definition and analysis activity embracing those activities shown on the chart.

We support the objectives of the Large Space Systems Technology Program and look forward to working with any of you, either individually or collectively, in making this Platform become a reality.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME: / DATE:
PROGRAM DEVELOPMENT	GEOSTATIONARY PLATFORM	CAREY JANUARY 1978

CONCLUSIONS

- GEOSTATIONARY PLATFORM HAS STRONG MISSION JUSTIFICATION, IS TECHNICALLY FEASIBLE, AND OFFERS MANY BENEFITS OVER THE TRADITIONAL SPECIALIZED SATELLITE APPROACH
- TECHNICAL CHALLENGES EXIST BUT NO SHOWSTOPPERS
- MSFC ACTIVITIES WILL CONTINUE:
  - FURTHER DEFINITION OF BASELINE AND ALTERNATIVES
  - BROADER USER BASE
  - FURTHER ESTABLISH SUPPORT REQUIREMENTS
  - COST AND COST COMPARISON
  - IDENTIFY SRT REQUIREMENTS AND INITIATE KEY ACTIVITIES
- CLOSE LIAISON REQUIRED WITH OAST LARGE SPACE SYSTEMS TECHNOLOGY PROGRAM

Figure 12

GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM

S. J. DENTON

MARSHALL SPACE FLIGHT CENTER

JANUARY, 1978

BRIEFING TOPICS

STRUCTURAL CONFIGURATION AND DESIGN CONCEPT
THERMAL CHARACTERISTICS
FLIGHT LOAD CONSIDERATIONS
ASSEMBLY APPROACHES

STRUCTURAL CONFIGURATION (Figure 1)

The Geostationary Platform configuration under consideration is shown in the following chart. A tetrahedron structural geometry was selected to provide inherent structural efficiency, all rigid members, and a compact arrangement of node points to which the antennae are mounted. RCS propulsion modules are located on outriggers at each of the four corners of the structural platform, and solar arrays are located at either end. The largest antenna, 30 meters in diameter, is centrally located to maintain symetry in weight distribution.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME:
PD 22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	S. DENTON
		DATE: JANUARY 1978

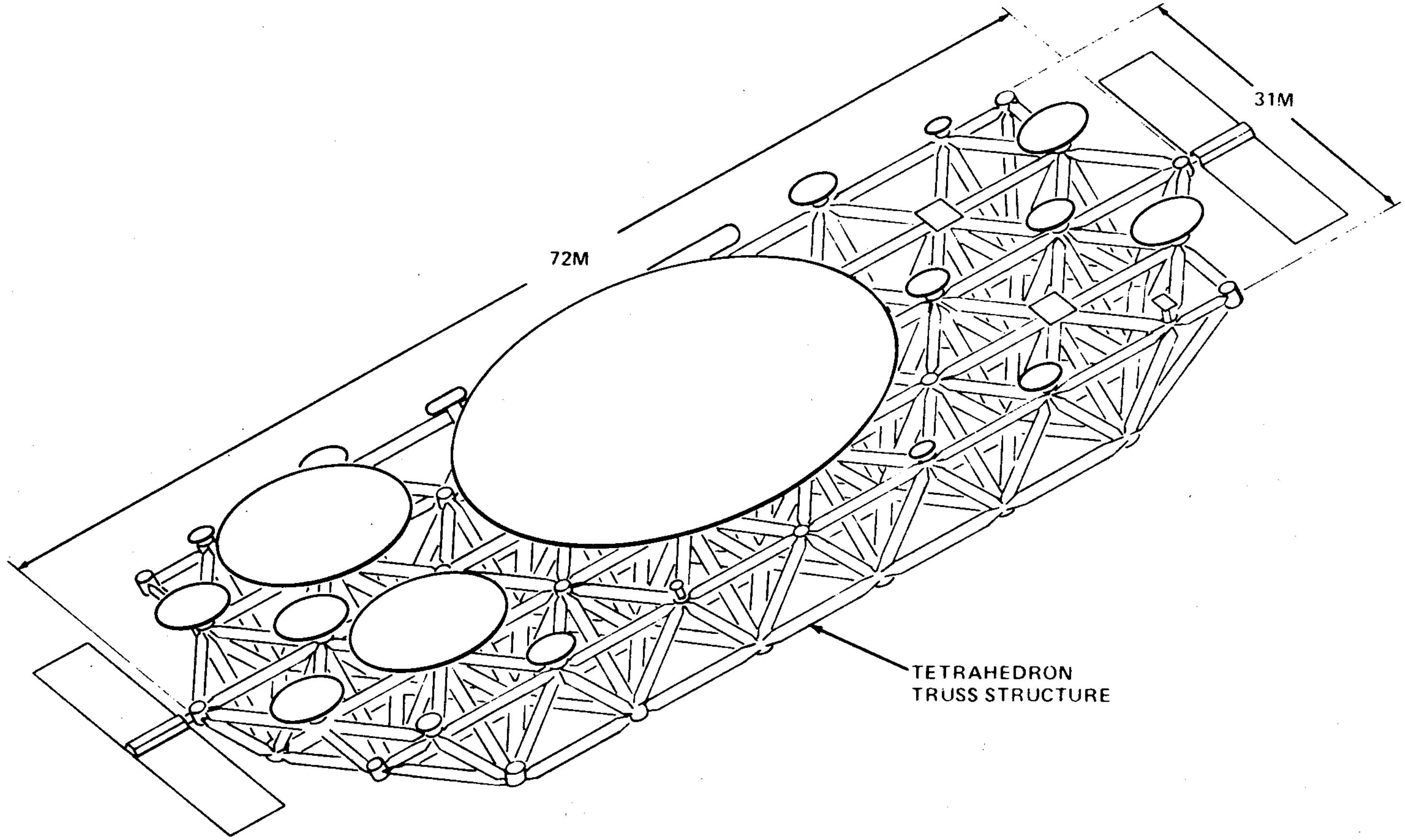


Figure 1

STRUCTURAL SYSTEM DESIGN OPTIONS (Figure 2)

An erectable structure design approach is currently preferred to a deployable system to minimize the number of STS launches required. Although a deployable structure would minimize the assembly tasks required, design concepts with competitive packaging densities and adequate member sizes are not currently available. A structural design approach, which utilizes the best features of both space fabricated and prefabricated structural members in an erectable system, is currently being evaluated.

ORGANIZATION		NAME
PD 22	**MARSHALL SPACE FLIGHT CENTER** GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	S. DENTON
		DATE JANUARY 1978

OPTIONS TO STRUCTURAL SYSTEM DESIGN APPROACH

ERECTABLE STRUCTURE

SPACE FABRICATED STRUCTURAL MEMBERS

- PROVIDES CAPABILITY TO PRODUCE LONG CONTINUOUS MEMBERS SIMPLIFYING ASSEMBLY

PREFABRICATED STRUCTURAL MEMBERS

- PROVIDES CAPABILITY TO TAILOR MEMBER SIZE AND MATERIAL TO SYSTEM REQUIREMENTS

STRUCTURAL DESIGN APPROACH CAN UTILIZE BEST FEATURES OF BOTH

DEPLOYABLE STRUCTURE

- MAJOR PORTION OF ASSEMBLY TASK ELIMINATED
- DEVELOPMENT OF CONCEPTS WHICH CAN BE PACKAGED IN CARGO BAY IS NEEDED

Figure 2

STRUCTURAL SYSTEM DESIGN CONCEPT (Figure 3, Figure 4)

The current structural system design approach for the Geostationary Platform is shown in the following two charts. Aluminum space fabricated beams, with a 1 meter cross-sectional depth, are utilized for the continuous longitudinal members. Graphite/epoxy prefabricated members, of the nested tapered tube type, are used for the interconnecting lateral and diagonal members. A node point spacing of 9 meters was selected to provide adequate separation of the various antennae mounted on the platform. In order to provide load paths aligned with the triangular cross section of the space-fabricated beams, the tetrahedron geometry deviates from an equilateral tetrahedron to one which is offset, having one member 7.8 meters in length.

ORGANIZATION	MARSHALL SPACE FLIGHT CENTER	NAME
PD 22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	S. DENTON
		DATE: JANUARY 1978

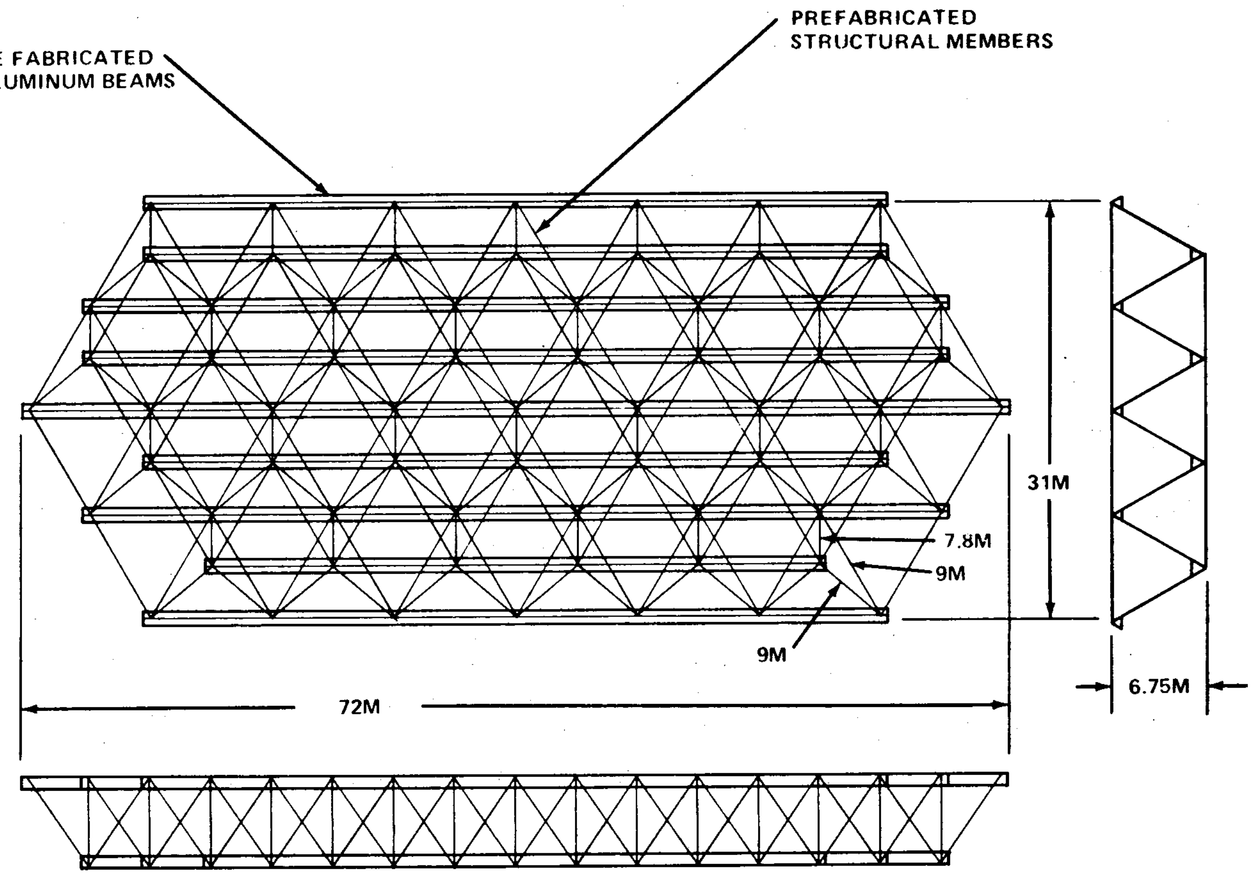


Figure 3

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME: S. DENTON
PD 22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	DATE: JANUARY 1978

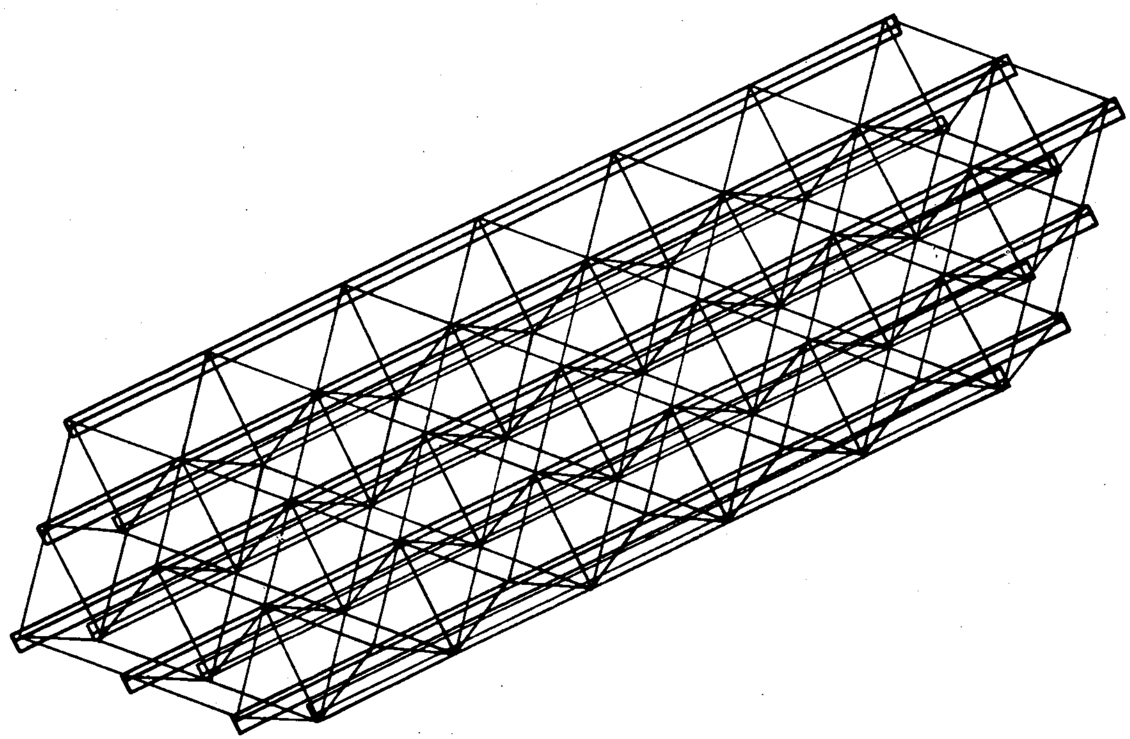


Figure 4

SPACE-FABRICATED BEAM DESIGN (Figure 5)

The design characteristics of the space fabricated beams utilized in the platform are shown. This design is taken from the Space Fabrication Demonstration System contract, currently in progress at Grumman Aerospace Corporation, which will produce a ground prototype metallic beam builder. The terminating tripod end fitting as shown would not be used for this platform structural concept.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME
PD22	ONE-METER BEAM DESIGN	S. DENTON
		DATE: JANUARY 1978

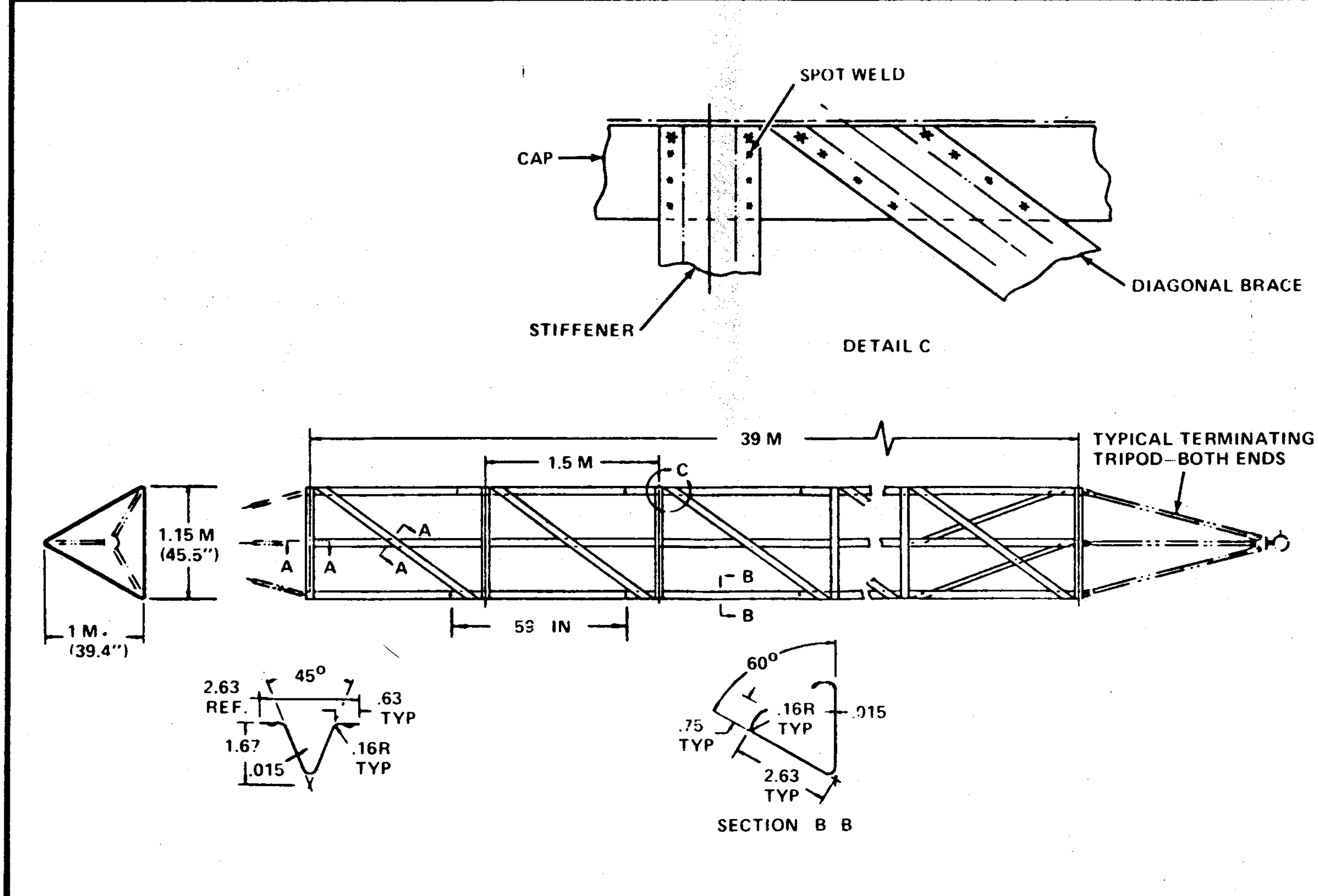


Figure 5

SPACE FABRICATED BEAM JOINT CONCEPT (Figure 6)

Since the space fabricated beams used in the platform structure are continuous in length, a joint concept is required to interconnect the lateral and diagonal members at each 9 meters of beam span. The saddle clamp joint concept shown on the following chart is a design approach to provide strut connections external to the 1 meter beam and to minimize the required volume of the joint. The joint indexes with the V-hat section of the beam lateral members and is installed in one piece. The joint also provides an interface for installation of an appropriate antenna support adapter. As a compromise to reduce the joint size, strut loads are introduced into the beam eccentric to the beam neutral axis. To introduce loads at the neutral axis, either an internal joint concept would be required, or one of excessive size.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME: S. DENTON
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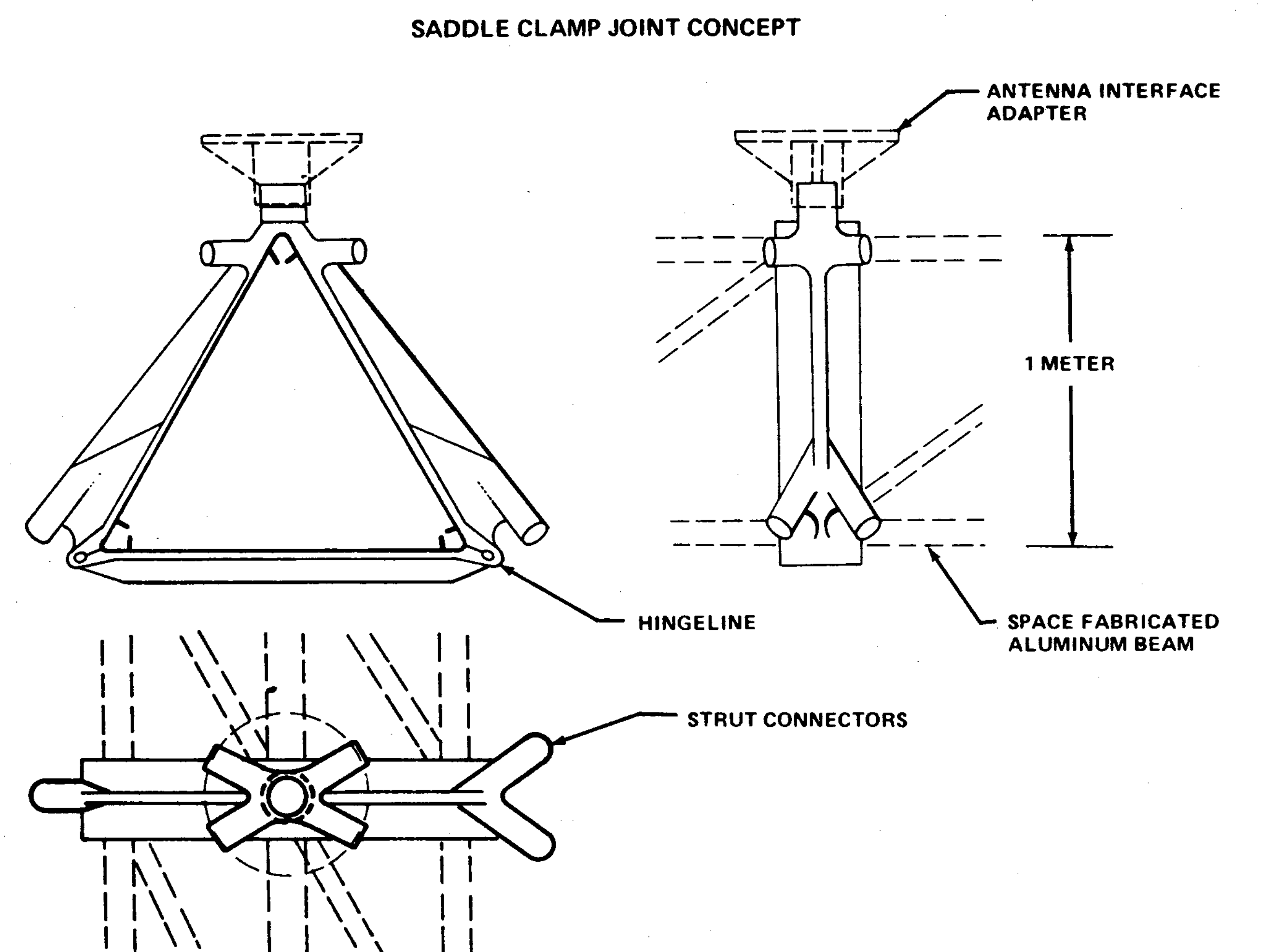

Figure 6

BEAM CAP MEMBER THERMAL CHARACTERISTICS (Figure 7)

The Geostationary Platform is oriented toward the Earth with its longitudinal axis perpendicular to the orbit plane, and since the space fabricated beams are parallel to this axis, they have similar thermal environments. The following chart indicates the temperature history of the cap members of the beams of the lower surface of the platform. The upper surface beams have a similar temperature history, but with a 4-hour phase shift due to a 30° difference in orientation to the sun. The cyclic change in temperature is due to the change in projected area of the cap member to the sun with change in orbit position. The peak temperature occurs with alignment of the open cavity of the cap member to the sun which has the effect of increasing absorptivity.

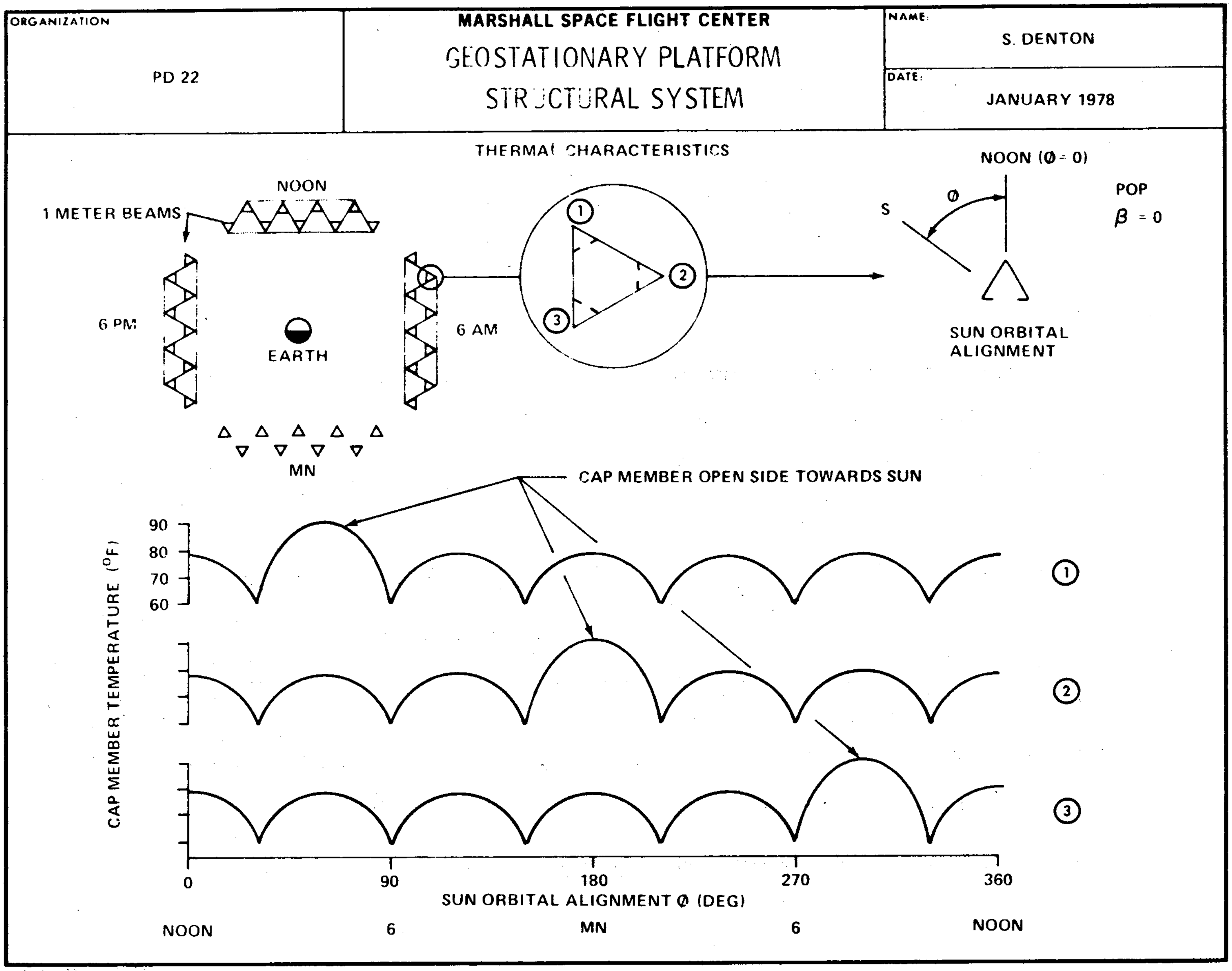
ORGANIZATION
PD 22
MARSHALL SPACE FLIGHT CENTER
GEOSTATIONARY PLATFORM
STRUCTURAL SYSTEM
NAME:
S. DENTON
DATE:
JANUARY 1978
THERMAL CHARACTERISTICS
NOON
1 METER BEAMS
6 PM
EARTH
6 AM
MN
1
2
3
NOON (Ø = 0)
S
Ø
POP
β = 0
SUN ORBITAL ALIGNMENT
CAP MEMBER OPEN SIDE TOWARDS SUN
CAP MEMBER TEMPERATURE (°F)
90
80
70
60
0
90
180
270
360
SUN ORBITAL ALIGNMENT Ø (DEG)
NOON
6
MN
6
NOON

Figure 7

BEAM CAP MEMBER SHADING EFFECTS (Figure 8)

Shading of the cap members of the space fabricated beam occurs when two members are co-aligned with the sun. This occurs twice per orbit for each cap member with a duration of 24.4 minutes. The following chart indicates the change in solar heat load during the shading period.

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THERMAL CHARACTERISTICS

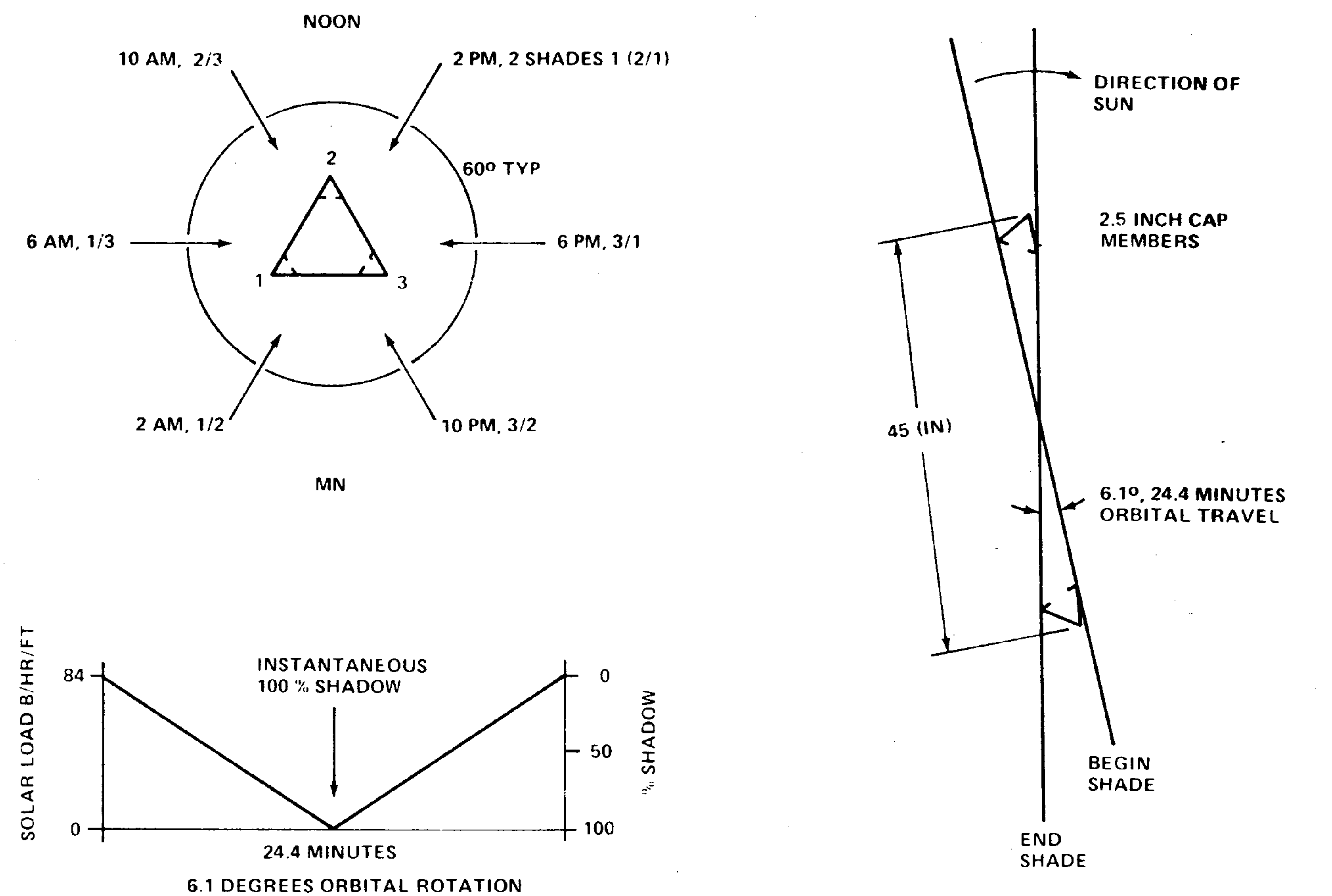

Figure 8

PLATFORM STRUCTURE TEMPERATURE HISTORY (Figure 9)

The following chart indicates the composite temperature history of the space fabricated beams contained in the platform structure. Also shown is the structural temperature change during maximum solar occultation duration, which occurs during the twice yearly equinox periods. These data are to be utilized to evaluate thermal distortion and thermal stress occurring in the structure.

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PD 22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	DATE: JANUARY 1978

THERMAL CHARACTERISTICS

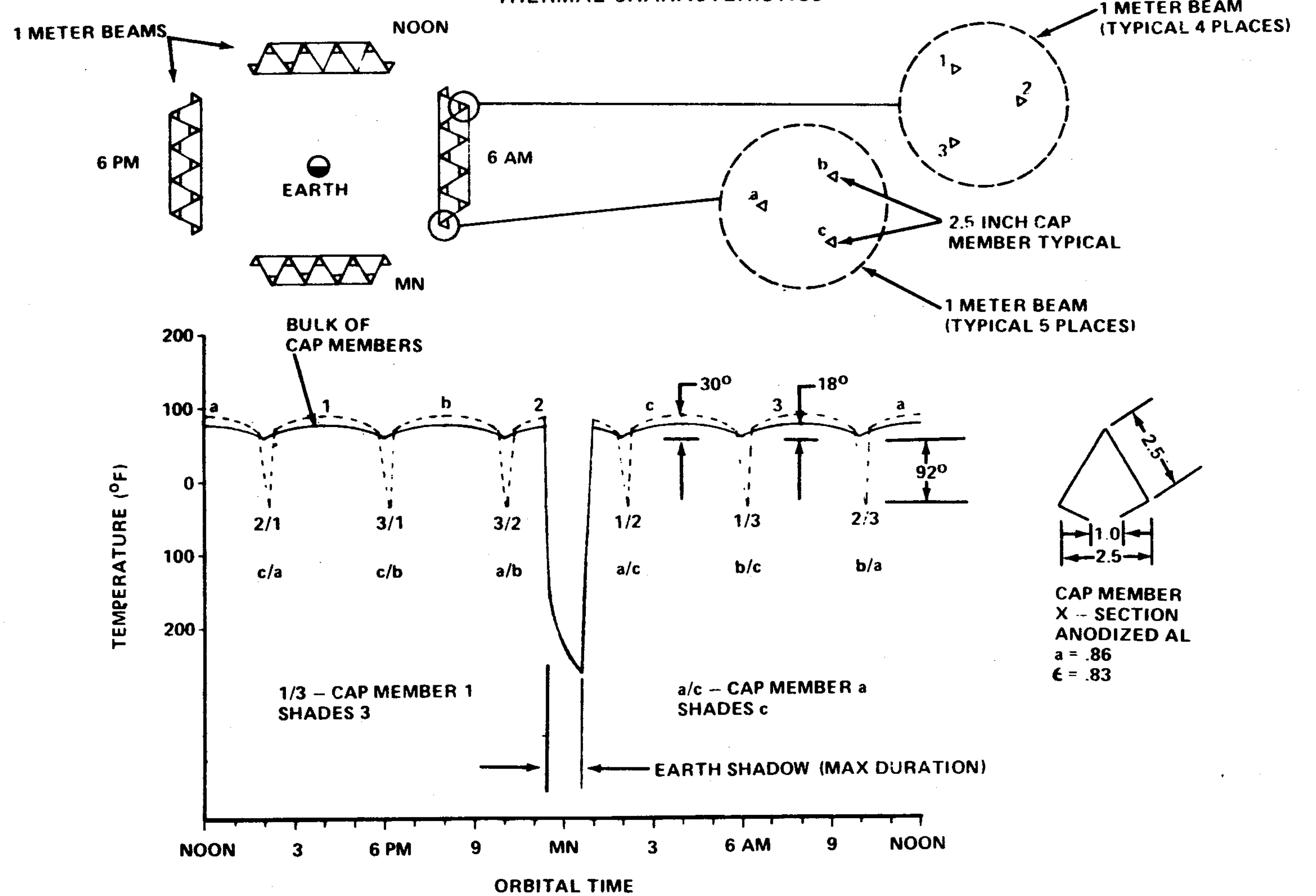

Figure 9

SEASONAL TEMPERATURE VARIATION (Figure 10)

Due to change in angle between the orbit plane and sun line (β) during the year and corresponding change in absorbed solar flux, the structural temperature will vary as indicated in the following chart. This has the effect of raising and lowering the temperature history as shown on the preceding chart.

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PD 22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	DATE: JANUARY 1978

THERMAL CHARACTERISTICS

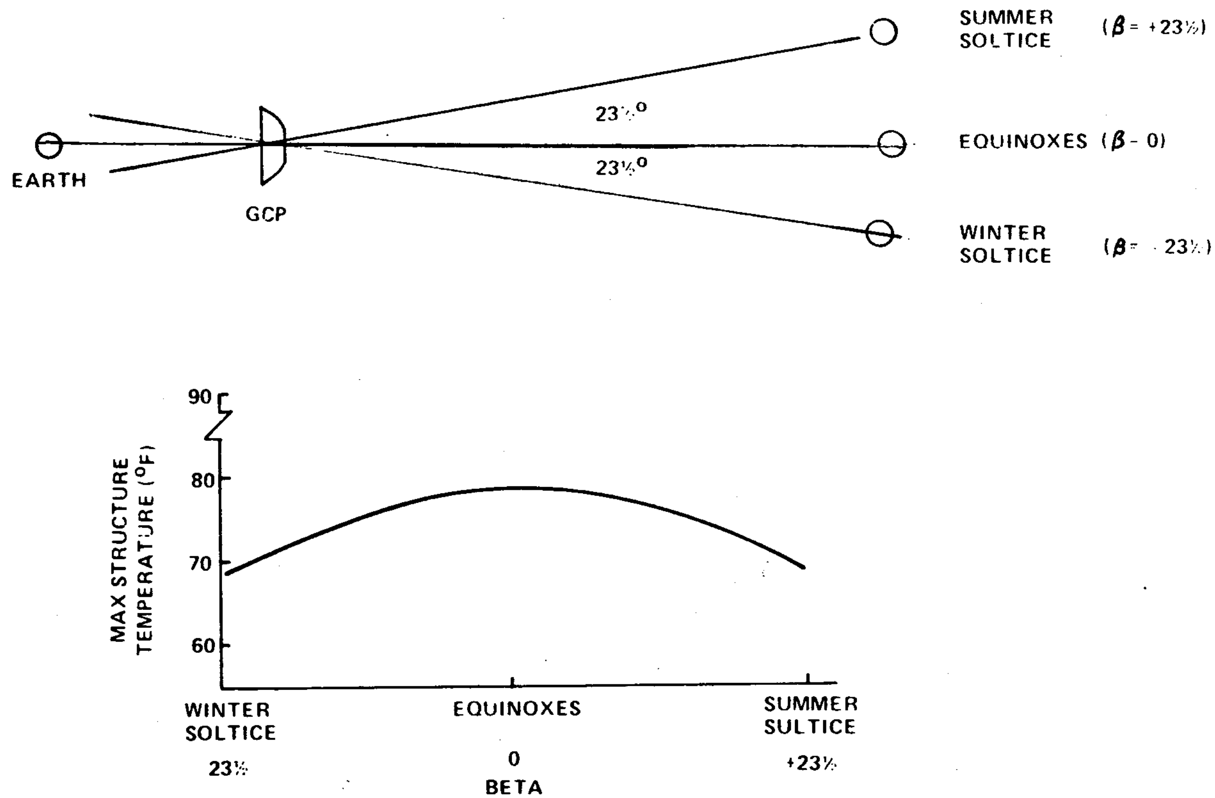

Figure 10

TRANSFER STAGE THRUST LOAD INTRODUCTION (Figure 11)

Options for the introduction of thrust loads into the platform structure are as indicated in the following chart. In each case, components which are not location critical, such as batteries and reaction wheels, are located in a subsystems module to preclude introducing their acceleration load into the platform structure. Although the centrally located thrust introduction point produces somewhat higher structural member loads than the other options, it is currently preferred due to additional design features made possible, such as providing a large rigid interface for the 30-meter-diameter antenna, and use as a support structure for component mounting during Orbiter flight.

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OPTIONS FOR INTRODUCTION OF THRUST LOAD

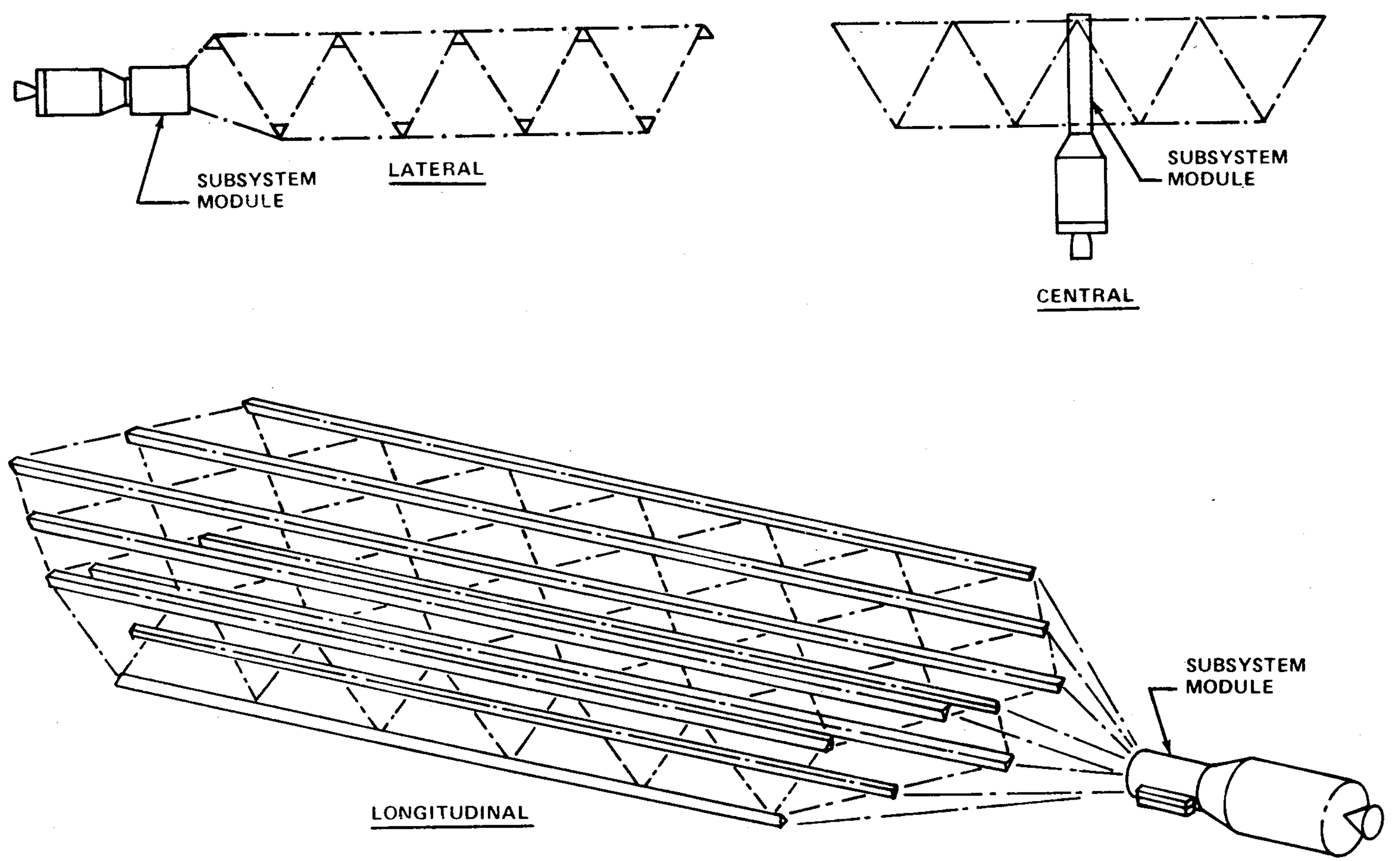

Figure 11

SUBSYSTEM MODULE DESIGN CONCEPT (Figure 12)

A design concept of a subsystem module which provides a docking interface for the OTV propulsion stage is shown. It interfaces with the structural members of the platform as shown, and provides a central location for initiation of structural assembly. As indicated, an interface for the 30-meter-diameter antenna is provided which can also be used for launch support of the antenna in its packaged form. The concept also provides the launch support structure for many of the platform components including prefabricated nested tapered tube structural members, RCS modules, and solar arrays. The internal volume can also potentially be utilized for additional component structural support during launch.

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SUBSYSTEM MODULE CONCEPT

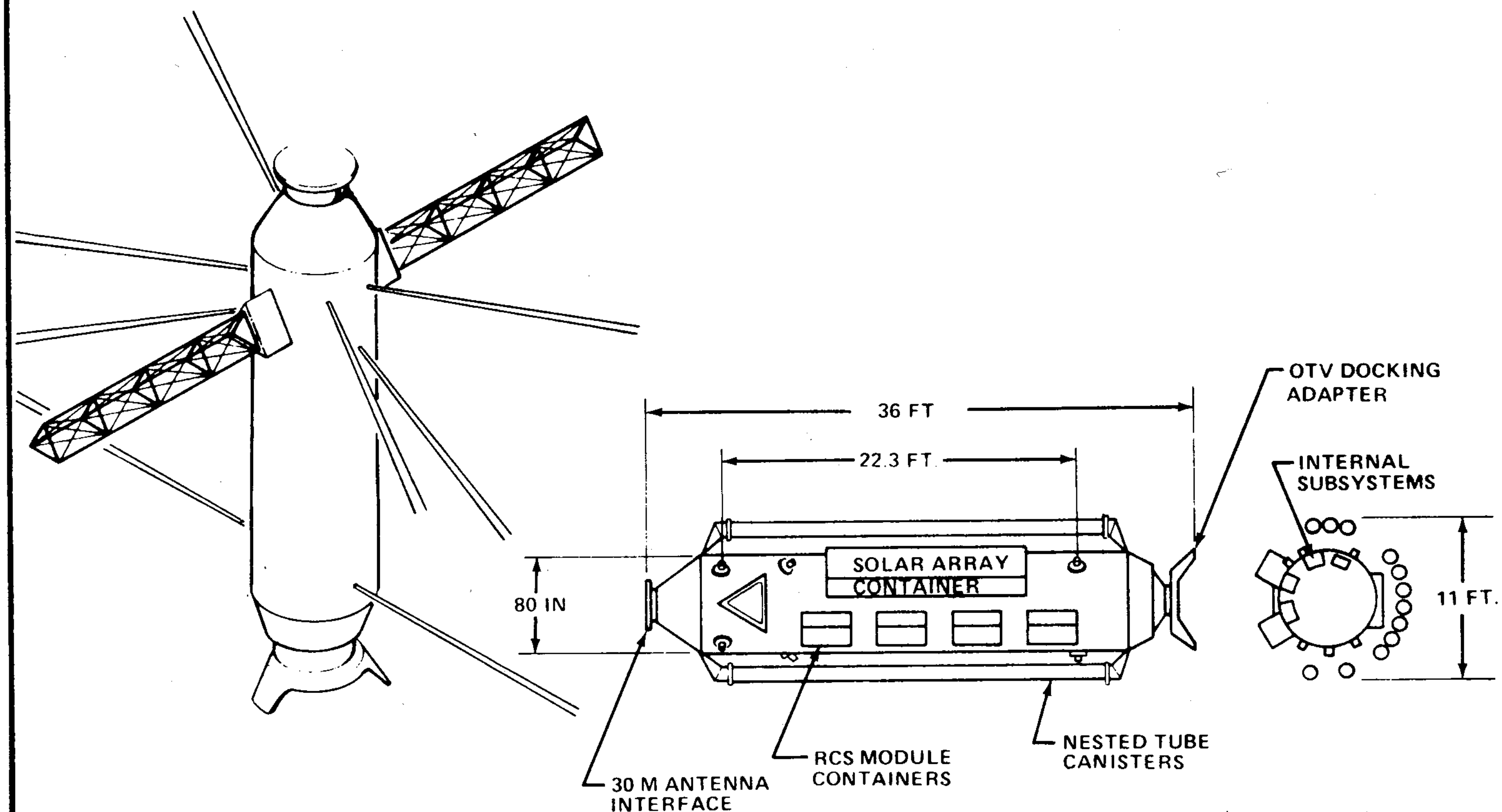

Figure 12

STRUCTURAL MEMBER LOADS DURING ORBIT TRANSFER (Figure 13)

Based upon OTV characteristics as indicated, resulting accelerations and corresponding average loads applied to the space fabricated longitudinal members are shown. Where a 50-percent engine throttle capability is utilized, applied loads exceed the capability of the current member design from the Grumman Aerospace "Space Fabrication Demonstration System" contract, which is used as a baseline.

ORGANIZATION:		NAME:
PD 22	**MARSHALL SPACE FLIGHT CENTER** GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	S. DENTON DATE: JANUARY 1978

THRUST EFFECT ON STRUCTURAL MEMBER LOAD

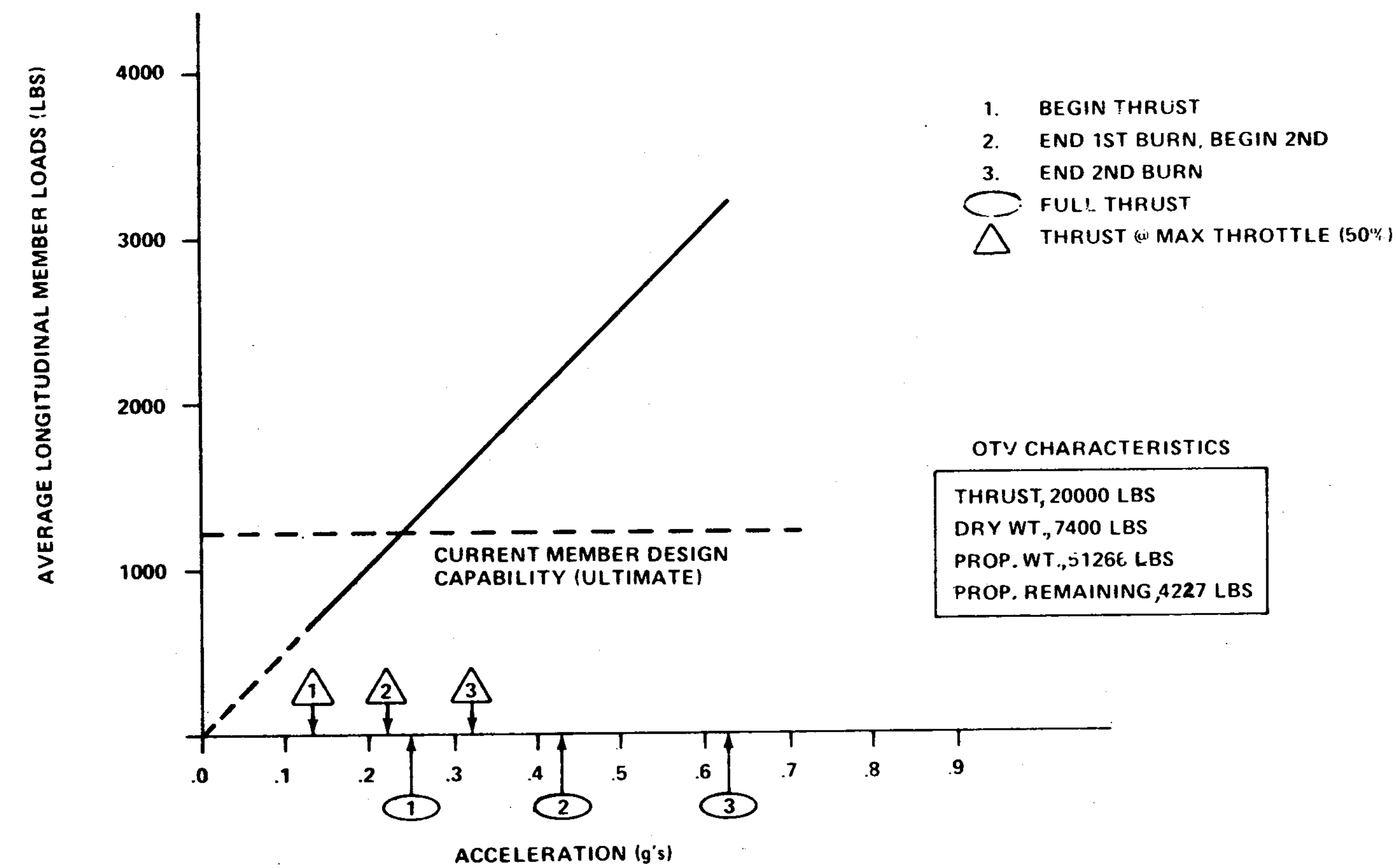

Figure 13

OPTIONS FOR STRUCTURAL LOAD COMPATIBILITY (Figure 14)

Potential options for meeting structural load requirements are as indicated. Unless structural weight becomes highly critical, the more desirable option would be to increase structural member load capability by increasing material thickness.

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PD22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	DATE: JAN 1978

OPTIONS FOR STRUCTURAL LOAD COMPATIBILITY WITH TRANSFER THRUST

OPTION	IMPACT
• INCREASE SPACE FABRICATED MEMBER LOAD CAPABILITY	• INCREASED STRUCTURAL WEIGHT
• INCREASE STRUCTURAL DEPTH	• INCREASED STRUCTURAL WEIGHT INCREASED ASSEMBLY COMPLEXITY
• LOWER THRUST OTV ENGINE (≈10K) WITH 50% THROTTLING	• DECREASED OTV MISSION FLEXIBILITY PERFORMANCE LOSS DUE TO LOW INITIAL ACCELERATION
• UTILIZE ALL PREFABRICATED MEMBERS SIZED FOR ADEQUATE LOAD CAPABILITY	• INCREASED ASSEMBLY COMPLEXITY

Figure 14

STRUCTURAL SYSTEM MEMBER REQUIREMENTS (Figure 15)

The required structural members of the platform system are summarized as indicated.

ORGANIZATION: PD 22	MARSHALL SPACE FLIGHT CENTER GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	NAME: S. DENTON DATE: JANUARY 1978

STRUCTURAL SYSTEM MEMBER REQUIREMENTS

		NUMBER	LENGTH – m
SPACE FABRICATED MEMBERS		3	63
		4	54
		1	45
		2	35
	TOTAL	10	520
PREFABRICATED MEMBERS		156	9.0
		28	7.8
	TOTAL	184	1622
JOINTS		67	SADDLE CLAMP WITH CONNECTIONS FOR 7 PREFABRICATED MEMBERS

Figure 15

ASSEMBLY-PREPARATION OF SPACE FABRICATED BEAM (Figure 16)

A major feature resulting from use of space fabricated beams is the capability to preassemble many of the system components, such as antennas, cables, and joints, to the beam prior to inclusion of the member into the structural assembly. By means of an erected work platform over the beam builder, a suitable astronaut work site can be provided for installation of these components, and eliminates the need for individual transport to the structural assembly area and any special equipment just for installation.

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PD22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	DATE: JANUARY 1978

PREPARATION OF SPACE FABRICATED BEAM PRIOR TO STRUCTURAL ASSEMBLY

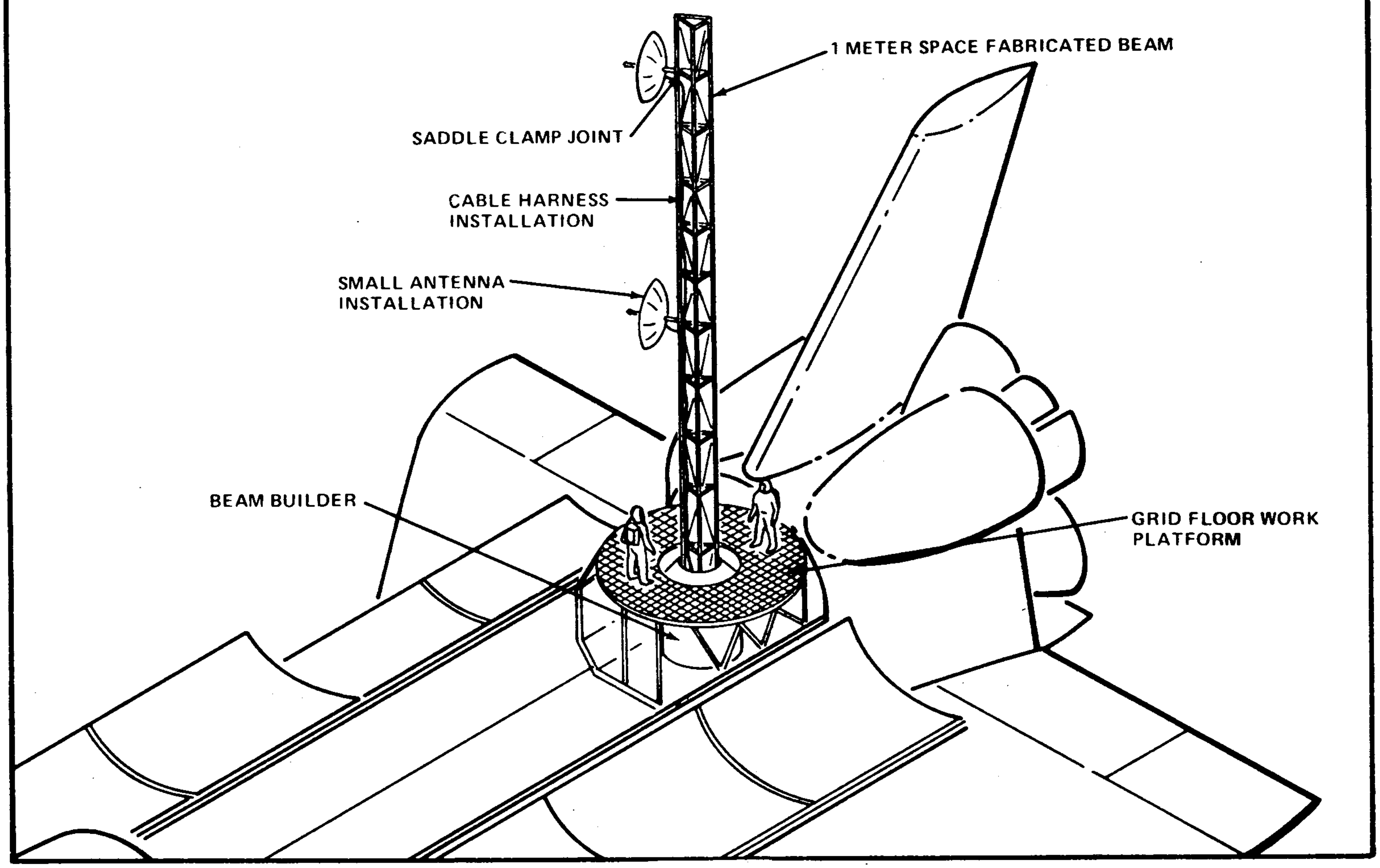


Figure 16

Construction Facility Concept (Figure 17, Figure 18)

The following two charts illustrate a typical concept of a construction facility which can be utilized for assembly of the Geostationary Platform. The concept is based upon a buildup of STS delivered modules and use of expended Shuttle External Tanks as a "strongback" platform. The concept would make major use of large ET mounted manipulator arms as the primary means of transport and positioning of components of the Geostationary Platform during assembly. The 25 kW Power Module is utilized to extend the Orbital stay time of the Orbiter and to provide power to the facility.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME: S. DENTON
PD22	CONSTRUCTION FACILITY CONCEPT	DATE: JANUARY 1978

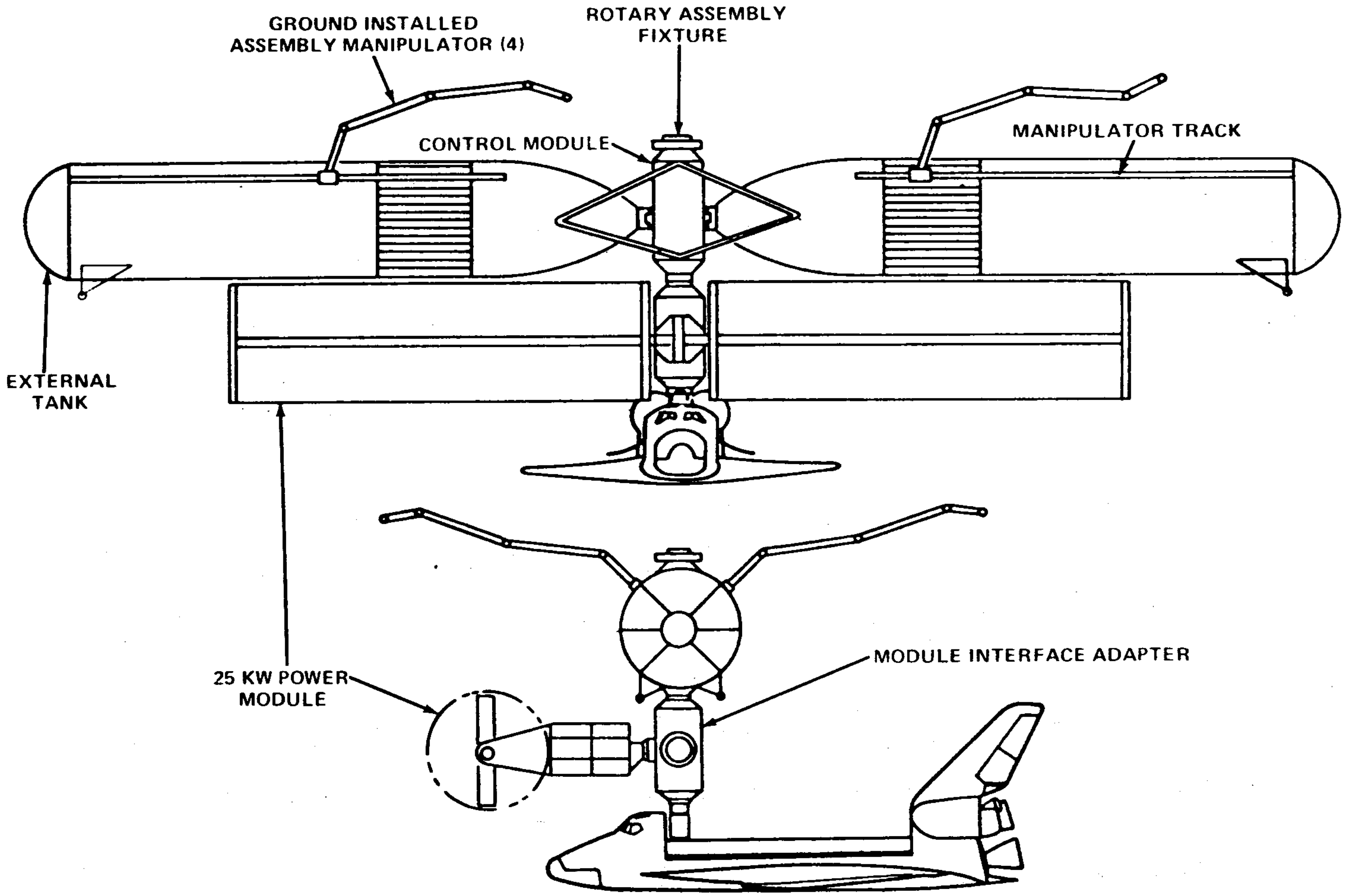


Figure 17

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME: S. DENTON
PD22	CONSTRUCTION FACILITY ASSEMBLY ENVELOPE	DATE: JANUARY 1978

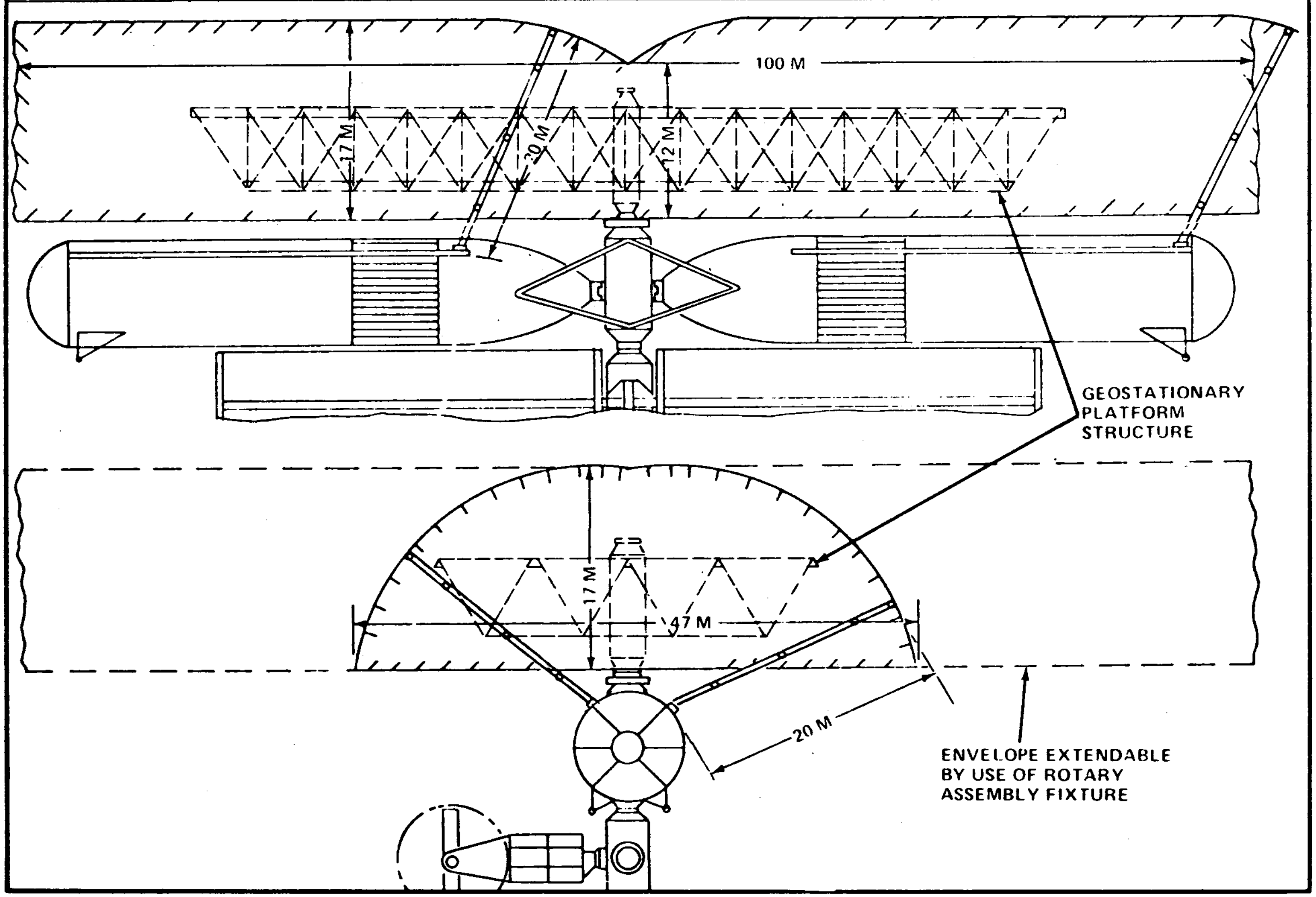


Figure 18

Platform Assembly From Orbiter **(Figure 19)**

The relatively modest size of the Geostationary Platform leads to the potential of assembly from the Orbiter with a minimum of support equipment. The following chart illustrates the relative size of the Geostationary Platform to the Orbiter and indicates a module that can be utilized to interface the 25 kW Power Module and provide a rotating support fixture for assembly of the platform. Required assembly equipment, needed for transport and positioning of members and components during assembly, are not shown and are subject to results of further studies involving comparative evaluation of capabilities utilizing EVA, teleoperator or manned maneuvering systems, and large manipulator arms.

ORGANIZATION	MARSHALL SPACE FLIGHT CENTER	NAME
PD22	GEOSTATIONARY PLATFORM STRUCTURAL SYSTEM	S. DENTON
		DATE: JANUARY 1978

PLATFORM ASSEMBLY FROM ORBITER

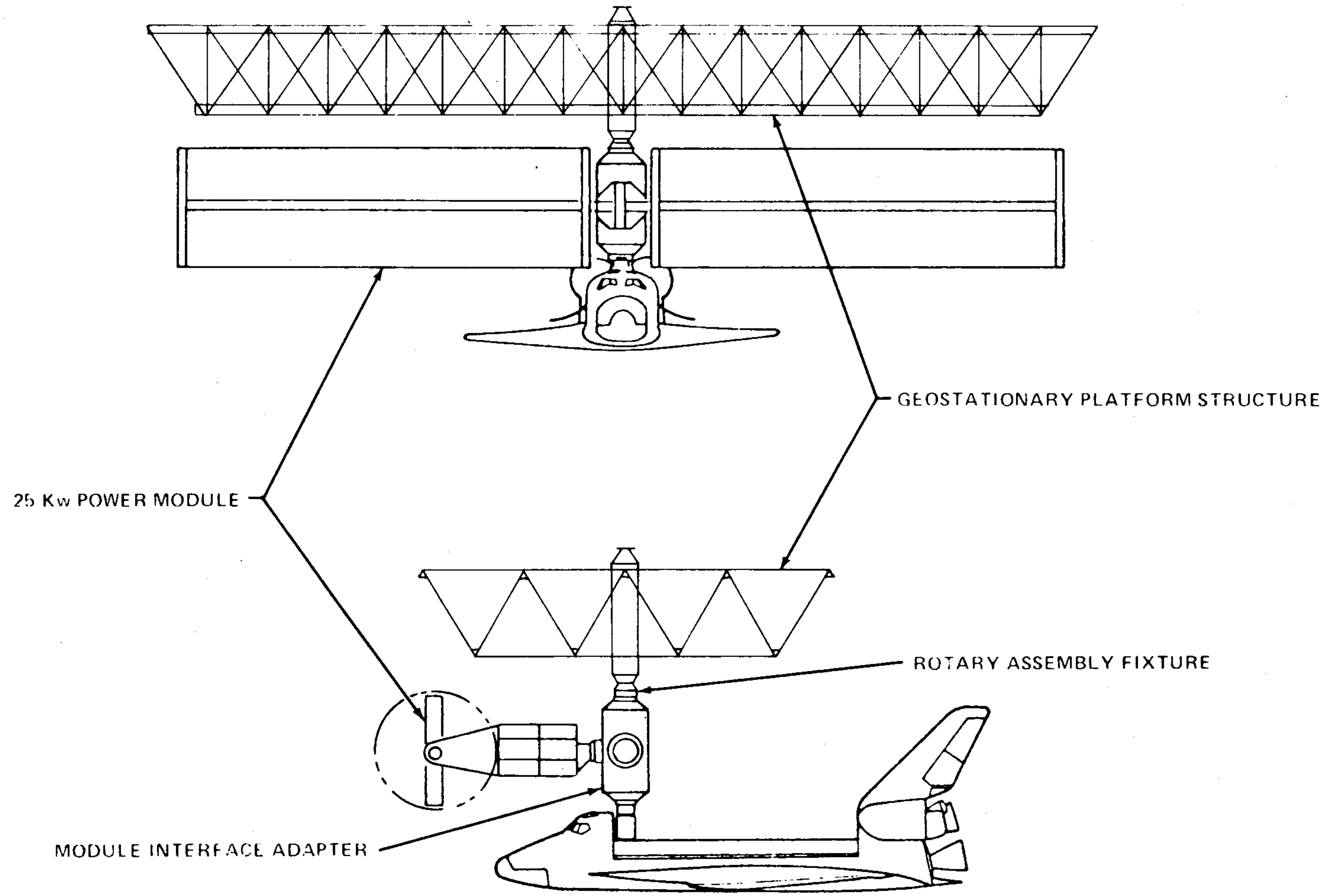
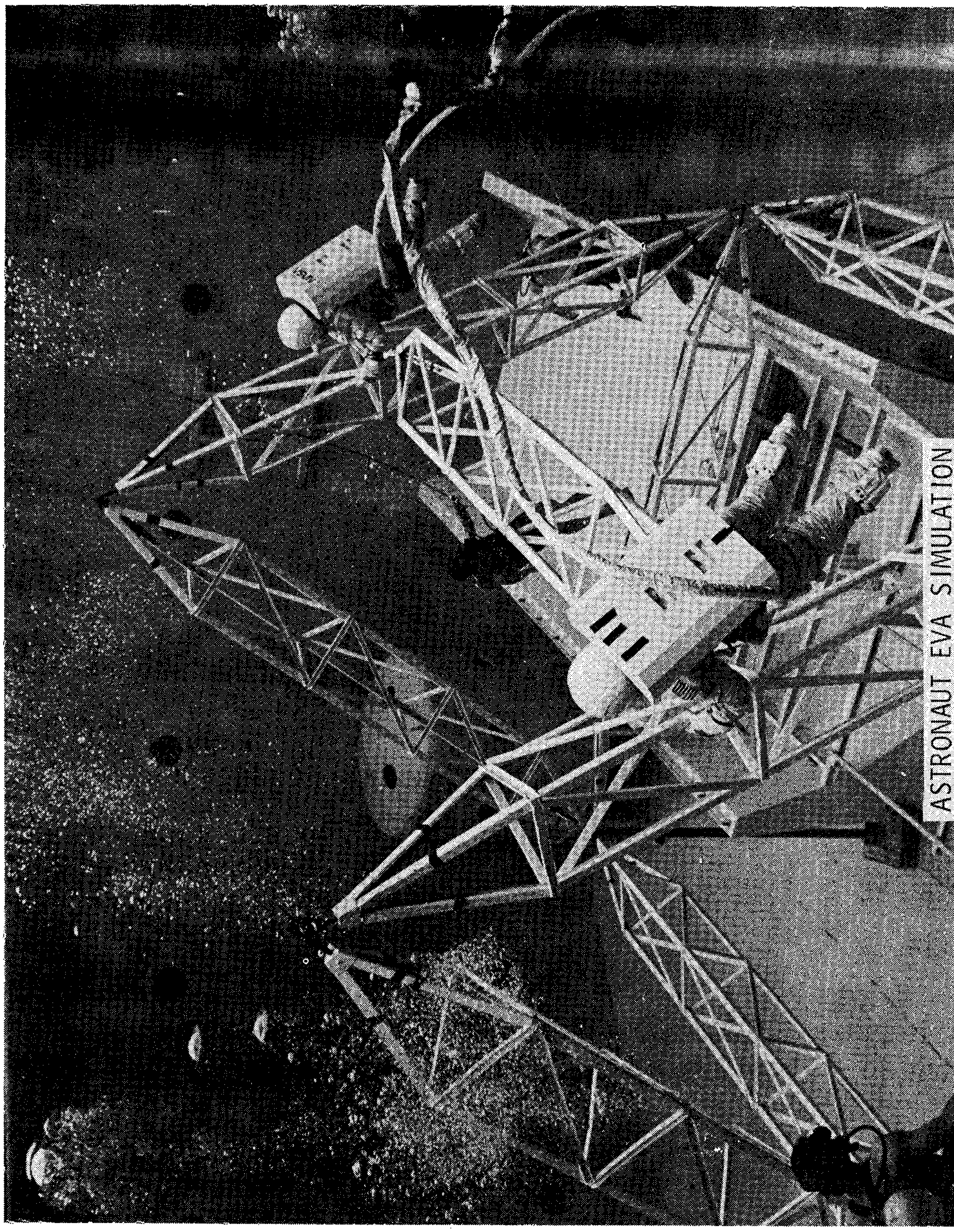


Figure 19

Neutral Buoyancy Tank Simulation of Assembly Tasks
(Figure 20; Figure 21; Figure 22)

MSFC has conducted several advanced studies over the past 3 years which involved the assembly of large space structures. Two questions which were considered high priority are related to transporting structural beams in space and man's ability to connect the beams together. MSFC initiated a test program in its neutral buoyancy tank last spring (1977) to develop answers to these and other fundamental assembly questions. The following three photographs are from tests conducted to evaluate the feasibility of moving beams by man, the Shuttle attached manipulator arm, and by a manned maneuvering system. It is interesting to note that, for the test article indicated, an average of approximately 10 minutes was required to manually transport, position, and attach each beam to another, by using a simple spring-loaded male/female connector.



ASTRONAUT EVA SIMULATION

Figure 20

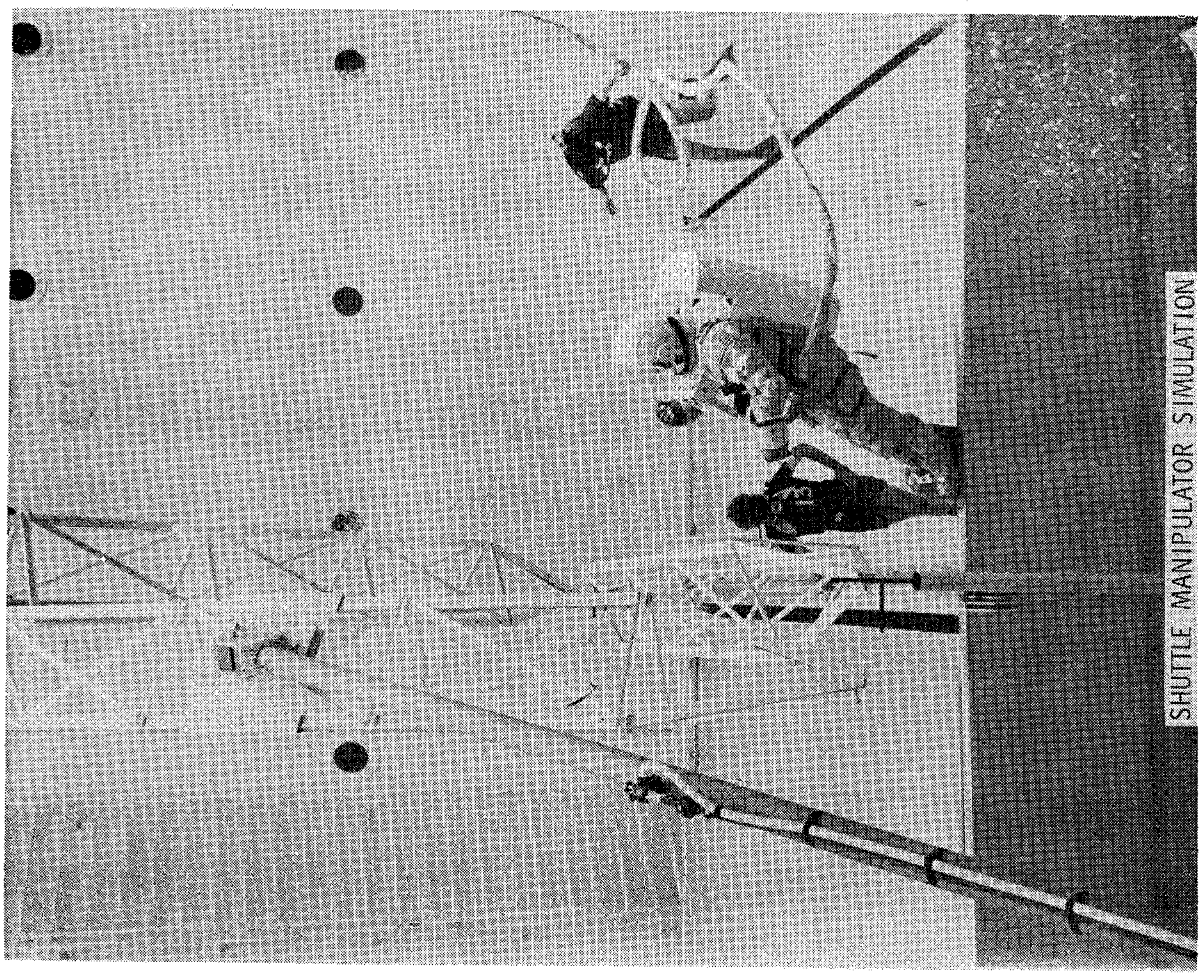


Figure 21

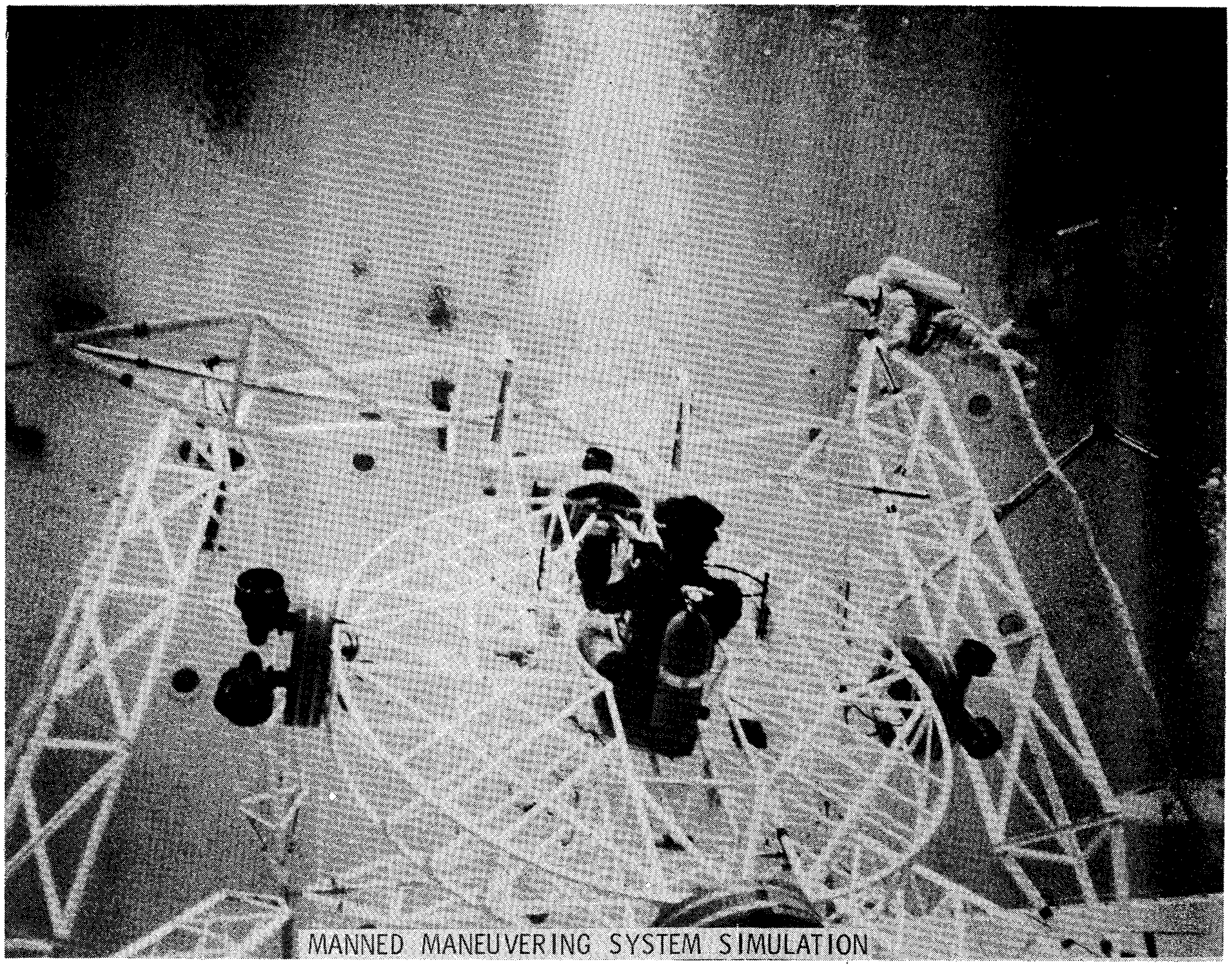


Figure 22

SYSTEM DYNAMICS AND SIMULATION OF LSS

R. F. RYAN

MARSHALL SPACE FLIGHT CENTER

JANUARY, 1978

SYSTEM DYNAMICS AND SIMULATION OF LSS (Figure 1)

Large Space Structures have many unique problems arising from mission objectives and the resulting configuration. Inherent in these configurations is a strong coupling among several of the designing disciplines. In particular, the coupling between structural dynamics and control is a key design consideration. The solution to these interactive problems requires efficient and accurate analysis, simulation and test techniques, and properly planned and conducted design trade studies. This paper deals with these subjects and concludes with a brief look at some NASA capabilities which can support these technology studies.

SYSTEM DYNAMICS AND SIMULATION OF LSS

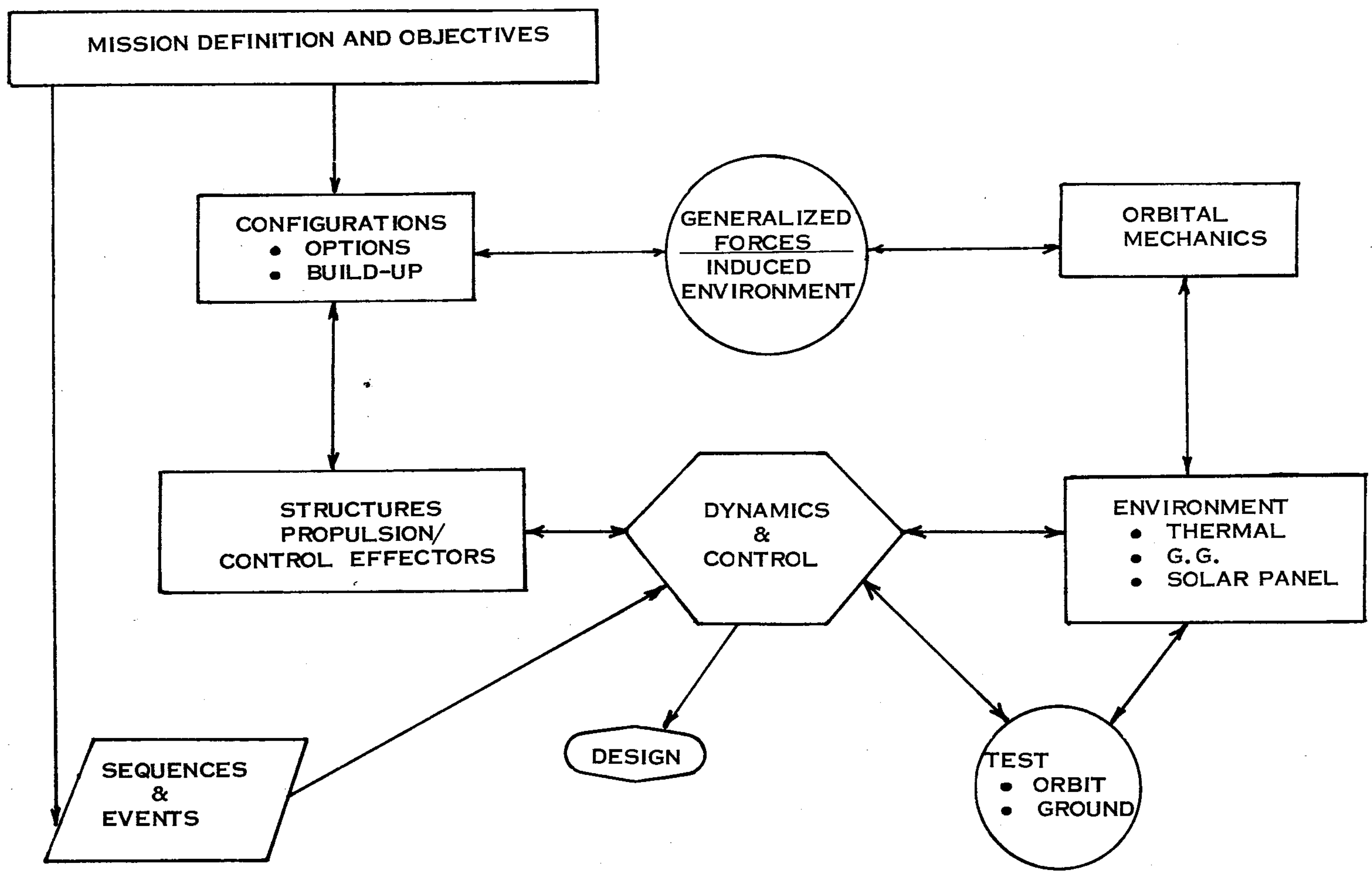
- CHARACTERISTICS OF SYSTEM DYNAMICS
- THE ROLE OF SIMULATIONS
- STATE-OF-THE-ART LIMITATIONS
- TECHNOLOGY REQUIREMENTS/PLANS
- NASA FACILITIES TO COMPLEMENT PLANS

Figure 1

SYSTEM DYNAMICS (Figure 2)

Experience during Skylab and now with Space Shuttle has taught the necessity of assuming a system focus during concept and design. In these two programs, there occurred a strong bonding among dynamics, control, aerodynamics, trajectories, and environment. In addition, large sensitivities in response occurred for small changes in system parameters. These are aggravated by the multibody configurations and nonlinearities arising from many joints. The same generic problems of multibody, multijointed, constraining control requirements and unique environments recur for LSS and demand the same system focus. This chart depicts the disciplines envisioned as being important in an interactive mode for LSS. Starting with the mission definition objectives, one can define configuration options and sequence of events, then proceed to conduct trade studies in order to understand the system and arrive at the optimum configuration and design. More than one configuration is required to achieve an optimized design since critical use must be made of configuration options, control system capabilities, materials, and manufacturing and assembly approaches. All arrows go in both directions depicting trade studies and feedbacks which are among two or more disciplines and within the total system to insure success. In summary, system analysis is a necessary focus for the technology phase to insure a successful program.

SYSTEM DYNAMICS



SYSTEM DYNAMICS DEFINED: THE CHARACTERIZATION OF A SPACE VEHICLE SYSTEM THROUGH SIMULATION OF DISCIPLINES THAT STRONGLY COUPLED WITH OR INFLUENCE THE DYNAMICS AND ASSOCIATED TRADE STUDIES NECESSARY FOR EFFICIENT DESIGN.

Figure 2

THE ROLE OF SIMULATORS (Figure 3)

The previous chart addressed the need for detailed systems analysis. This chart addresses the roles of simulation, analysis, and test. Simulation of the system, as discussed previously, is one key to proper design. It is the tool that can efficiently accomplish the inter-discipline trades required. Analysis supports the simulation by providing input check cases for the simulation and trends. Experiences on Skylab and Shuttle have demonstrated the merit of using very simplified analysis to bracket parameters and show trends, using the complex simulations for detail trades and design. The incorporation of springs and dampers between the support efforts and simulation was deliberate to demonstrate that this support must be a dynamic interaction with damped characteristics. This implies information feed forward and back. Environment was separated from test and analysis for emphasis, even though it splits between the two. The accurate prediction of critical environments is mandatory for unconservative design. Test is very important in terms of input data and verification. Although no interaction is shown between analysis and test, obviously there is one. With Large Space Structures, there exists a complex trade required among control system complexity, accuracy of structural characterization, analysis, and test requirements.

A brief look has been taken at the role of the different aspects of system dynamics in development of the LSS program. The next two charts address state-of-the-art limitations and LSS technology needs in these areas.

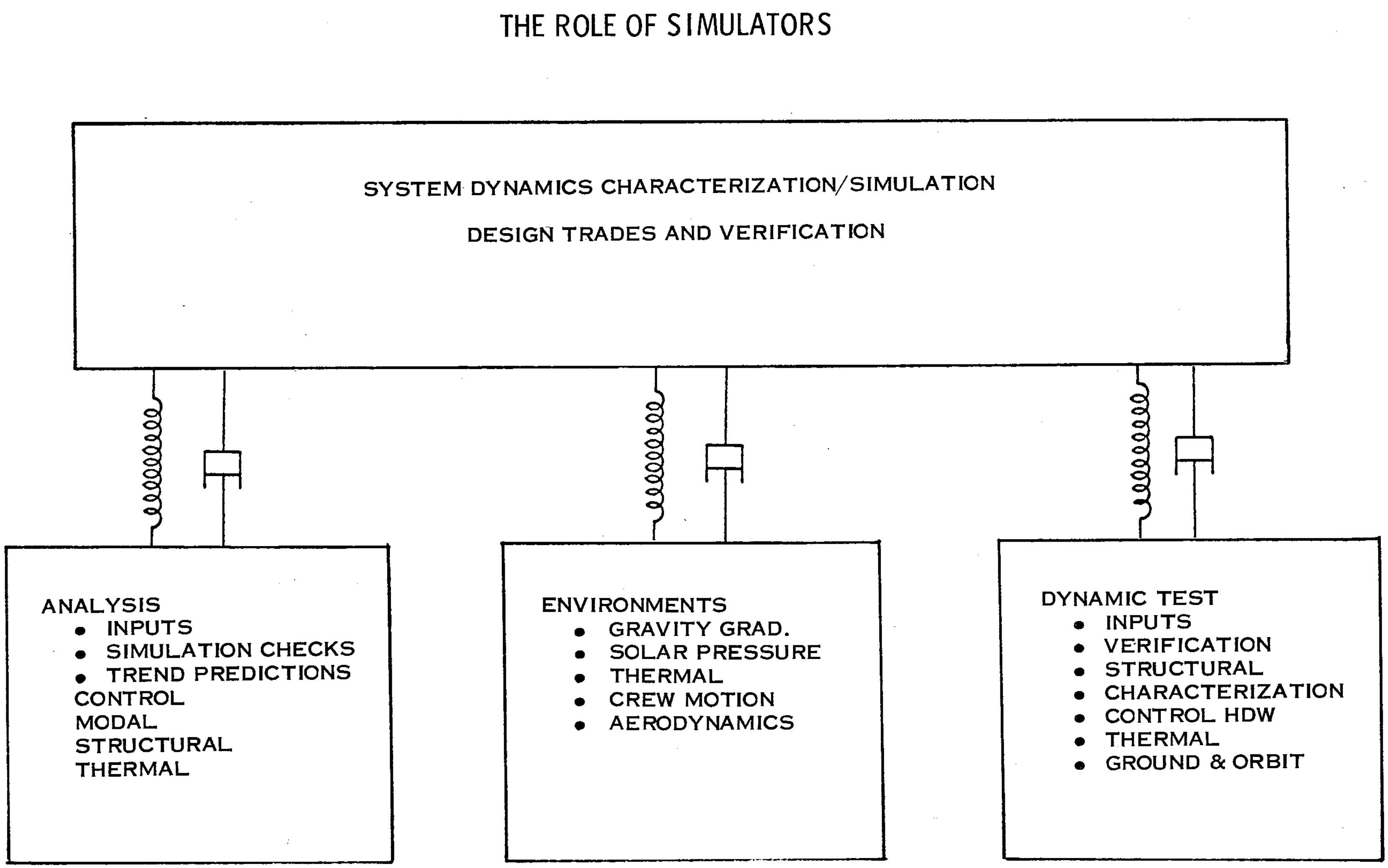

THE ROLE OF SIMULATORS
SYSTEM DYNAMICS CHARACTERIZATION/SIMULATION
DESIGN TRADES AND VERIFICATION
ANALYSIS
• INPUTS
• SIMULATION CHECKS
• TREND PREDICTIONS
CONTROL
MODAL
STRUCTURAL
THERMAL
ENVIRONMENTS
• GRAVITY GRAD.
• SOLAR PRESSURE
• THERMAL
• CREW MOTION
• AERODYNAMICS
DYNAMIC TEST
• INPUTS
• VERIFICATION
• STRUCTURAL
• CHARACTERIZATION
• CONTROL HDW
• THERMAL
• GROUND & ORBIT

Figure 3

STATE-OF-THE-ART LIMITATIONS (Figure 4)

Limitations in the state-of-the-art technology in the areas of analysis, test, and simulations are well known. A partial summary of these limitations is shown on this chart. Basically, analysis is limited to linear approaches or very simplified nonlinear ones. Test is basically high g with size limitations and inefficient data acquisition and evaluation approaches. Simulations have been very useful but are limited in the degrees of freedom that can be simulated. Facilities cost can be excessive unless all the stops are pulled in terms of ingenuity and skills. Large space structure technology requirements follow.

STATE-OF-THE-ART LIMITATIONS

SIMULATIONS:

- LIMITED DEGREES OF FREEDOM
- BASICALLY ONE G
- FACILITIES/COST
- INEFFICIENT WITHOUT ANALYSIS SUPPORT FOR TREND

ANALYSIS:

- STRUCTURAL INTERACTION
  - DIGITAL CONTROL SYSTEMS (FIXED SINGLE SAMPLE RATE) AND CONTINUOUS CONTROL SYSTEM
  - PARTIAL STATE FEEDBACK, ANALOG, AND DIGITAL FILTERS
  - PRIMARILY LINEAR
  - LIMITED DEGREES OF FREEDOM
- STRUCTURAL DYNAMICS
  - PRIMARILY LINEAR
  - FINITE ELEMENT
  - MODAL SYNTHESIS
  - SUBSTRUCTURING

TEST:

- ONE G EXCEPT SMALL SCALE MODELS
- PRIMARILY GROUND TEST
- EXCITATION POINT FORCE SHAKER
- TIME CONSUMING
  - RANDOM EXCITATION (EXCESSIVE EVALUATION)
  - MODAL EXCITATION (EXCESSIVE TEST TIME)
- TEST/ANALYSIS CORRELATION INEFFICIENT, MAN-IN-LOOP JUDGMENT

Figure 4

LSST REQUIREMENTS (Figure 5)

Large Space Structures, or more precisely the different programs or uses of large space structures in space, levy a unique set of requirements on design and, therefore, on technology. Not only must some configurations have specific orientations in space; but, in addition, their shape must be controlled. The structure must be assembled or manufactured in space or both. This leads to growth accommodation requirements, joints, and various roles of man manipulator interactions. Size limits ground testing as do design requirements that are stiffness-driven instead of strength-driven. Digital control systems need the fullest exploitation to lessen the structural design impacts and reduce the need for development of specific materials.

Large Space Structures technology must develop simulations that are large-scale, nonlinear, time-scaled with growth potential. This is not only important for design, but also for realtime support during buildup and operations. Skylab demonstrated this through the use of a time-scaled Skylab orbit simulation that included dynamics and control to plan practically daily the most optimum maneuvers for experiments in terms of fuel usage (RCS propellant). In addition, simulations are needed for optimal design approaches, man loop interaction with system and closed loop control, and special trade studies. In addition to the items discussed, the development of good simulations requires the development of vehicle performance criteria. The developments must continue toward simplifying the simulations without losing any essential characteristics.

In the area of analysis, techniques for analysis using all the uniqueness of digital control systems are needed (e.g., multi-sample rate, variable skip, and nonlinear filtering). The old problem of state estimation is with us and has even more importance in Large Space Structures without detailed all-up dynamic test verification. Testing is a real problem. The low g environment coupled with the structure size basically eliminate ground testing. Some means must be devised to couple limited ground testing (component and scaled) with on orbit testing and analysis in an optimum way as a verification tool.

Backup charts are provided which delineate the state-of-the-art limitations and LSST requirements as they relate to various future programs. The first backup chart is flight regimes versus disciplines and covers all aspects from ground up and back. A generic notation is used to show gaps. The second chart shows the breakdown in much greater detail in terms of discipline or discipline subsets versus various concepts of Large Space Structures. The last chart provides the same generic information for control technology.

LSST REQUIREMENTS

SIMULATION:

- LARGE SCALE NONLINEAR TIME VARYING, TIME-SCALED SIMULATION INCLUDING ORBITAL MECHANICS, CONTROL, THERMAL, DYNAMICS, ENVIRONMENTS
  - TRUNCATION CRITERIA/APPROACHES
  - IDEALIZATION CRITERIA
  - PERFORMANCE CRITERIA
- OPTIMAL DESIGN APPROACHES
- MAN-IN-LOOP CONTROL/CLOSED LOOP CONTROL
- DISTRIBUTED SENSORS AND ACTUATORS (EFFECTORS)

ANALYSIS:

- CONTROL
  - MULTI-SAMPLE RATE, VARIABLE SKIP, DIGITAL CONTROL SYSTEM ANALYSIS, AND DESIGN APPROACHES
  - GROWTH ACCOMMODATING
  - MULTI-VARIABLE CONTROL TECHNIQUES
  - PARAMETER ESTIMATION METHODS
  - DISTURBANCE ACCOMMODATING CONTROLLER
- COMBINED ANALYSIS/TEST VERIFICATION APPROACH
- DETAILED DYNAMIC MODELS
  - NONLINEAR ELEMENTS COMBINED WITH LINEAR ELEMENTS
  - JOINTS, MACRO ELEMENTS, ETC.
  - IDEALIZATION CRITERIA
  - TRUNCATION CRITERIA
  - DECOMPOSITION CRITERIA

TEST:

- LOW G SIMULATION
- ON ORBIT TESTING APPROACHES
- GROUND TESTING APPROACHES, ELEMENTS, SCALE
- IMPROVED/OPTIMIZED EXCITATION, ACQUISITION, REDUCTION, EVALUATION, AND CORRELATION TECHNIQUES

Figure 5

KEY ISSUES (Figure 6)

This chart lists some of the key issues in various disciplines important to system dynamics and the associated trade studies. The listing is not intended to be all inclusive and is biased by the author's experience. Major issues occur in each discipline area as well as among the disciplines (e.g., in the integrated dynamics area, key issues involving test and analysis roles and the resulting technologies as discussed previously). How to model and simulate nonlinearities is a key area, as well as whether to design for stiffness requirements structurally or depend on control systems to provide the equivalent stiffness. The source for control authority is very important as is the sensor choice, location, and control logic. In the area of design criteria, the choice of unconservative approaches for parameter variations and methods of combining these in design studies is necessary if low cost/high reliability are to be achieved. Other key issues deal with choice of materials, role of man-in-the-loop, verification approaches for models, and the role of on-orbit test, control system update, etc., versus all-encompassing ground test and development. The approach of desensitizing the system to variations of system parameters versus brute force design approaches could lead to efficiency and cost savings. Based on these issues and the LSS technology requirements discussed previously, a plan of attack is now developed.

KEY LSST ISSUES, SYSTEM DYNAMICS

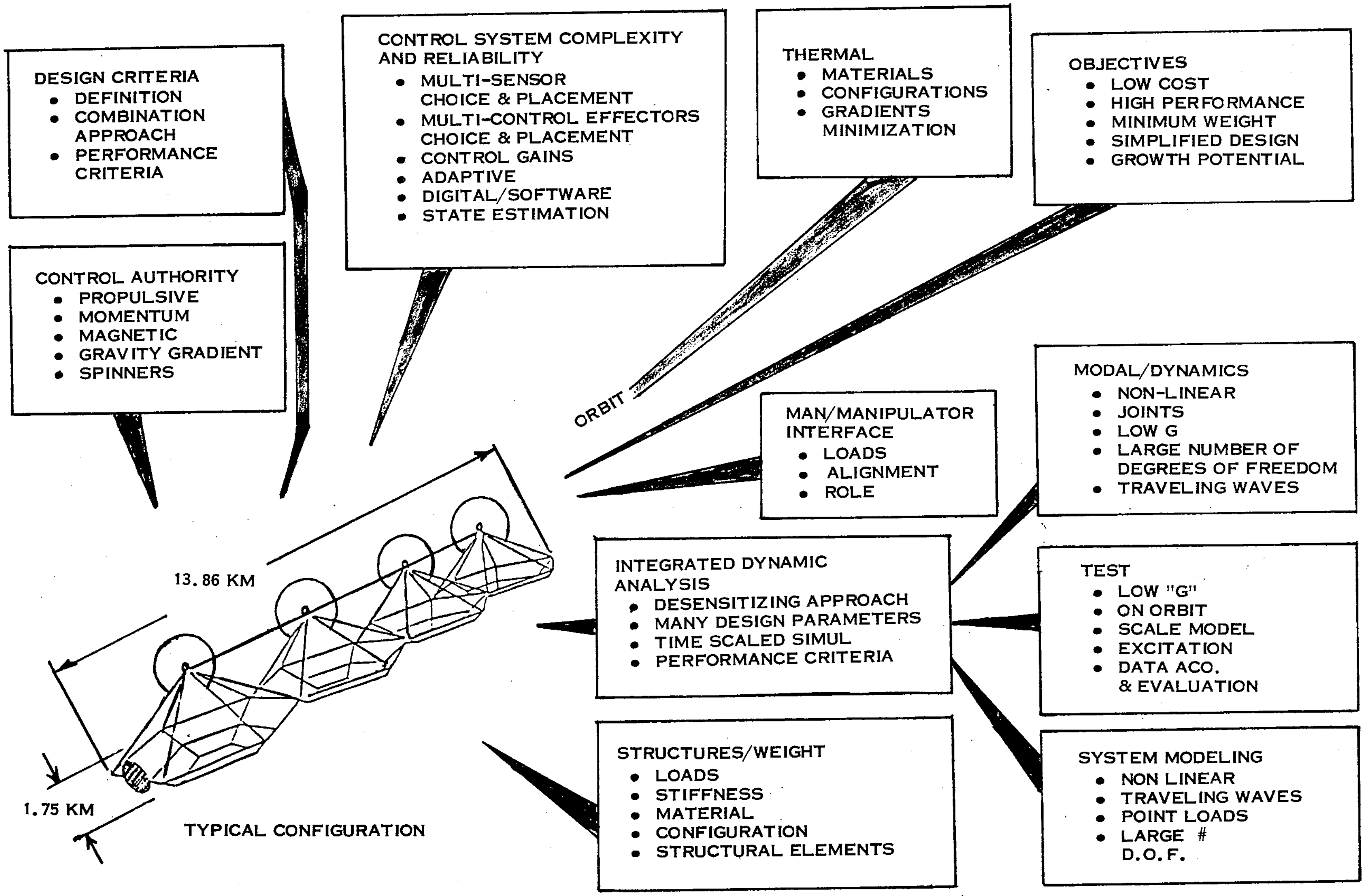


Figure 6

TECHNOLOGY PLANS (Figure 7)

This chart breaks out the six major tasks required to develop appropriate technology for LSS. Emphasis is placed in the dynamic and control areas where NASA has unique facilities and experience. Tasks that arise in other areas were not addressed due to the author's limited time and may be just as important. Neither do these omissions signify that NASA does not have other unique skill areas important to LSS. This program as outlined would provide the technology for ferreting out the basic characteristics, in a dynamic and control sense, of LSS and for optimal design and verification. A look is taken next at some of the NASA facilities readily adaptable to support this technology plan.

TECHNOLOGY PLANS

- DETERMINE THE LIMITATIONS OF CURRENT STATE-OF-THE-ART DYNAMIC ANALYSIS APPROACHES FOR CURRENT LSST CONFIGURATIONS THROUGH DESIGNING, ANALYZING, AND TESTING (SCALE MODEL) ONE SYNTHESIZED CONFIGURATION (INCLUDING FLIGHT EXPERIMENTS).
- DEVELOP AN OPTIMIZED TEST/ANALYSIS APPROACH MAKING MAXIMUM USE OF NASA UNIQUE FACILITIES.
- DEVELOP A MULTI-DISCIPLINED OPTIMIZED DESIGN APPROACH WHICH PROPERLY WEIGHS OR TRADES THE VARIOUS ASPECTS REQUIRED FOR LOW COST, HIGH PERFORMANCE SPACE STRUCTURES.
- DEVELOP DETAILED SYSTEM SIMULATIONS USING NONLINEAR AND LINEAR ELEMENTS COUPLED TOGETHER AND EXISTING NASA CAPABILITIES IN ORDER TO EFFICIENTLY CONDUCT KEY TRADE STUDIES AND EVALUATE MAN-IN-THE-LOOP ROLES.
- DEVELOP AND VALIDATE EFFICIENT AND ACCURATE ANALYSIS TECHNIQUES FOR DYNAMIC MODELS, DIGITAL CONTROL SYSTEMS, DISTRIBUTIVE CONTROL SYSTEMS, SPIN STABILIZATION, SYSTEM MODELS, AND INCORPORATION OF SYSTEM PARAMETER VARIATIONS.
- DEVELOP CONTROL TECHNOLOGY IN TERMS OF EFFECTOR CHOICE, SENSOR CHOICE, CONTROL LOGIC, DIGITAL CONTROLLERS, SOFTWARE, AND STATE ESTIMATION.

Figure 7

SIMULATION (Figure 8)

This chart is a layout of one of NASA's simulation setups to support docking type dynamic analysis. The point with this chart is to show the components and their flexibility and not the particular application. Key elements are the hybrid computer which simulates all dynamics and control not simulated with hardware and acquires and reduces all data as well as the six degrees of freedom motion base which can carry the docking mechanism for contact dynamics or simulate a moving vehicle. Visual displays are shown with control system sensor interfaces, etc. Finally, the man-in-the-loop control panels, etc., are illustrated. In addition, there is available a T-27 (cockpit) with visual displays, flat air bearing table, buoyancy simulator, etc.

SIMULATION DIVISON - BLDG. 4663

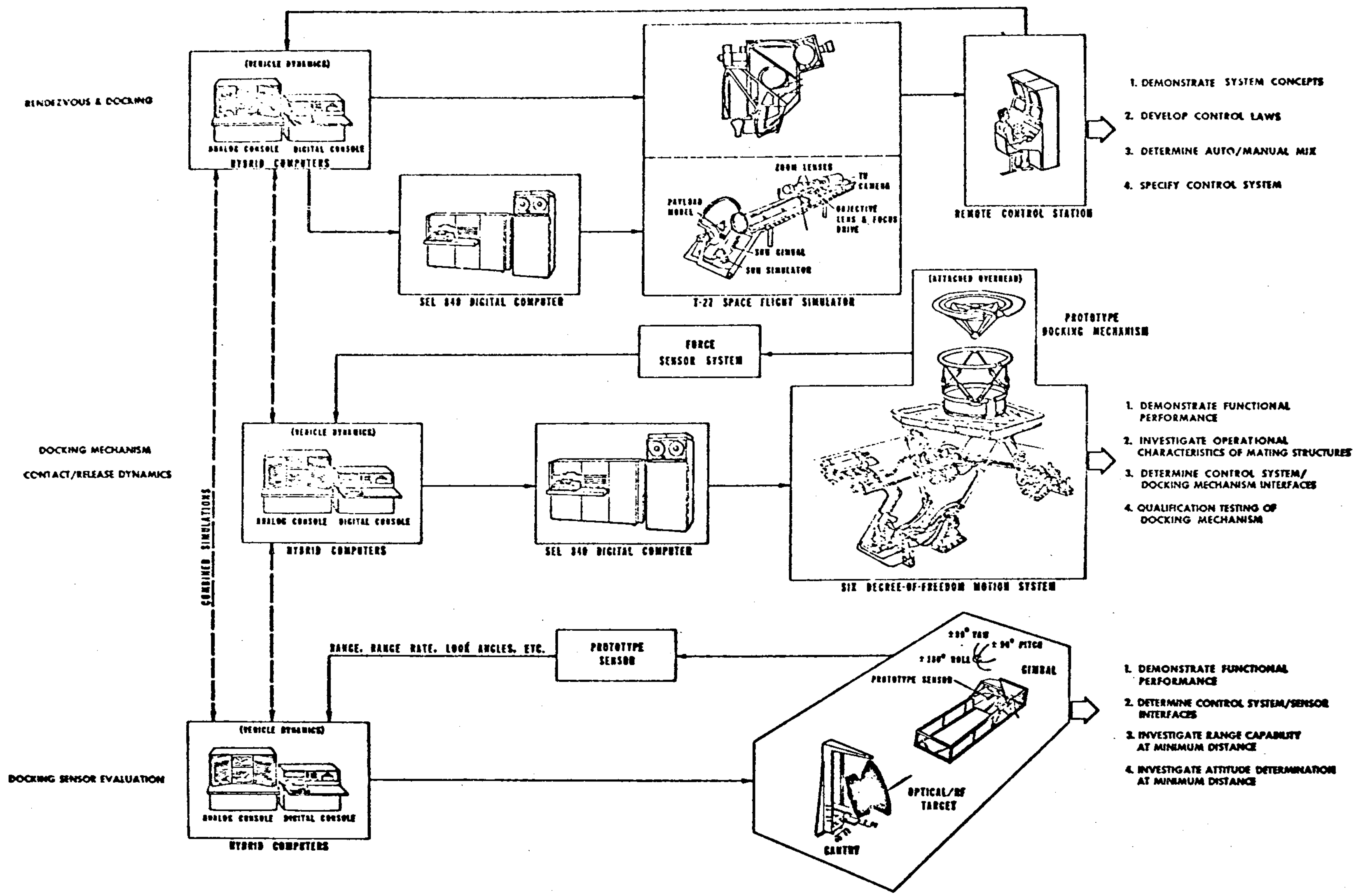
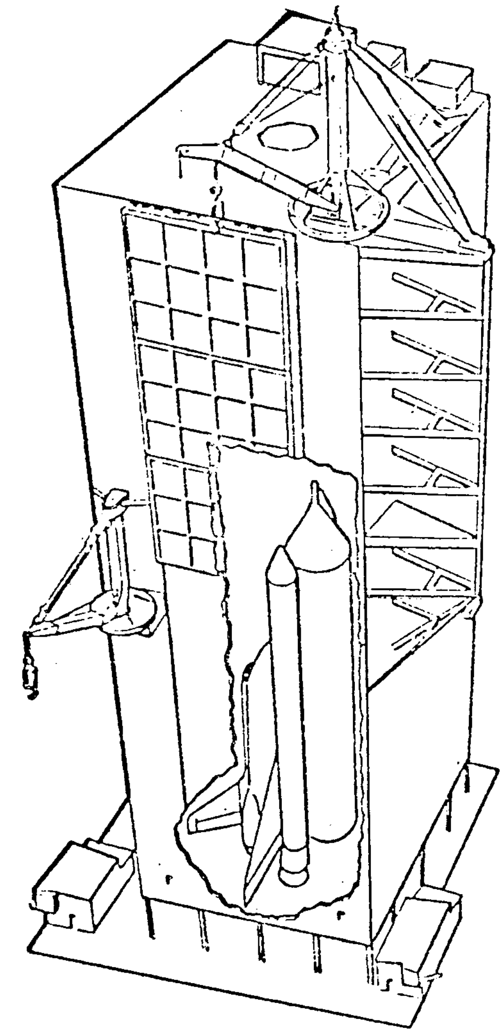


Figure 8

TESTING (Figure 9 - Figure 13)

All aerospace programs to date have used full scale dynamic tests to verify dynamic models used in design and verification. This chart shows the Saturn dynamic test stand modified to handle the full scale dynamic test of Space Shuttle in its two configurations. This test is to be conducted this year. The facility, with its support equipment, can be modified for many LSS tests. There exist small dynamic test facilities, such as the one presently being used for Shuttle lox tank modal survey test, and structural test facilities, currently in use to verify the Shuttle External Tank design.



SHUTTLE CONFIGURATION

MATED VEHICLE GROUND VIBRATION TEST

(MVGVT)

Figure 9

FUTURE TECHNOLOGY AND DISCIPLINE REQUIREMENTS

FLIGHT REGIMES	ACOUSTIC/ OVERPRESSURE	AEROELASTIC	VIBRATION MODEL	DYNAMIC TEST	LOADS AND RESPONSES	OPTIMIZATION COUPLING
LAUNCH						
• LIFTOFF	G	E	E	G	G	G
• ASCENT	E	E	E	G	G	G
ORBIT						
• OPERATIONS	E	-	G	G	G	G
• DEPLOYMENT	-	-	G	G	G	G
• REBOOST	E	-	G	G	G	G
REENTRY AND RECOVERY	G	G	G	-	G	G
LARGE SPACE STRUCTURES		ENVIRONMENT PREDICTION				
• TRANSPORTATION		T	G	T	T	-
• MANUFACTURING AND ASSEMBLY		T	G	T	T	T
• SHAPE AND POINTING (MISSIONS)		T	G	T	T	T
• REBOOST		G	G	G	G	G

E - EXPERIENCE G - EXPERIENCE WITH GAPS T - NEW TECHNOLOGY

Figure 10

STRUCTURAL DYNAMICS TECHNOLOGY STATUS SUMMARY

PRESENT CAPABILITY	TECHNOLOGY REQUIRED	PROJECT: ERECTABLE STRUCTURES	DEPLOYABLE PLATFORMS	DEPLOYABLE REFLECTORS	SPACE BASED POWER	506 RTOPS	LARGE LIFT VEHICLES
COMPUTATIONAL							
-STATIC	-HYBRID, DIRECT	X	X	X	X		X
-EIGENVALUE	INTEGRATION/MODAL						
-SUBSTRUCTURE	-CONSTRAINTS & LINKS	X	X	X	X		X
-MODAL SYNTHESIS	-MAN-IN-LOOP SIMULATIONS	X	X	X	X		
-COMPLEX EIGNEVALUE	-INTERACTIVE MODES	X	X	X	X		X
-HYBRID & DIGITAL	-TRANSIENT & DOCKING	X	X	X	X		
SIMULATIONS							
-NONLINEAR DOCKING							
OPTIMIZED TEST ANALYSIS							
-MOUSE	-SCALE MODEL	X	X	X	X		X
	-EXCITATION	X	X	X	X		X
	-ACQUISITION/EVALUATION	X	X	X	X		X
	-ON ORBIT	X	X	X	X		X
OPTIMIZATION							
-LAUNCH VEHICLE	-LOADS/CONTROL/CONFIGU-						
CONTROL/LOADS/	RATION/MATERIALS	X	X	X	X		X
PERFORMANCE							

Figure 11

STRUCTURAL DYNAMICS TECHNOLOGY STATUS SUMMARY (CONT'D)

PRESENT CAPABILITY	TECHNOLOGY REQUIRED	PROJECT: ERECTABLE STRUCTURES	DEPLOYABLE PLATFORMS	DEPLOYABLE REFLECTORS	SPACE BASED POWER	506 RTOPS	LARGE LIFT VEHICLES
ELEMENTS (SPAR)							
-FLAT PLATE	-SOLID HYBRID					X	X
-STRAIGHT BEAM	-ELEMENT OF REVOLUTION	X	X	X	X	X	X
-SHELL OF REVOLUTION	-CURVED BEAM, SHELL	X	X	X	X	X	X
-FLUID	-FLUID FLOW						X
-MATRIX	-MACRO	X	X	X	X	X	X
	-JOINTS & FILLETS	X	X	X	X	X	X
AMPLITUDE & RESPONSE							
-MODAL/ENVIRONMENT	-MATERIAL NONLINEAR	X	X	X	X		
MULTIBODY	-GEOMETRIC NONLINEAR	X	X	X	X		
MULTICOAST	-DIRECT INTEGRATION	X	X	X	X		
DIGITAL	-WARPED ELEMENTS	X	X	X	X	X	
HYBRID	-TRAVELING WAVES	X	X	X	X		
-DIFFERENTIAL STIFFNESS							

Figure 12

ORBITING LARGE SPACE STRUCTURES CONTROL TECHNOLOGY REQUIRED

CONTROL TASKS	CENTRALIZED/ DECENTRALIZED CONTROL	MULTIVARIABLE CONTROL TECHNIQUES	GROWTH ACCOMMODATING CONTROL	SENSOR/ACTUATOR CONCEPTS	LARGE SCALE ANALYSIS/SYNTHESIS METHODS	SPIN STABILIZATION	OPTIMAL STRUCTURE/CONTROL DESIGN	ACTIVE BEAM/TRUSS CONFIGURATION DESIGN	EXPANDED DISTURBANCE MODELS	DISTRIBUTED SYSTEM CONTROL	DOCKING CONTROL ALLOCATION COLLISION AVOIDANCE	MANIPULATOR CONTROL TECHNOLOGY	PARAMETER ESTIMATION METHODS	ADVANCED DIGITAL DESIGN TECHNIQUES	CONTROL SYSTEM DYNAMIC INTER-ACTION ISOLATION
FABRICATION		IN PLACE		NEW ITEM				NEW ITEM	IN PLACE						
ASSEMBLY	IN WORK	NEW ITEM	IN WORK	IN PLACE	NEW ITEM				IN PLACE		IN WORK				IN WORK
TELEOPERATORS/FREE FLYERS/MANIPULATORS		IN PLACE		IN PLACE	IN PLACE							NEW ITEM		NEW ITEM	NEW ITEM
ON-ORBIT DYNAMIC TESTING		IN PLACE		IN WORK	IN PLACE		NEW ITEM		NEW ITEM	NEW ITEM			NEW ITEM	NEW ITEM	IN WORK
TRANSPORTATION		NEW ITEM			NEW ITEM		NEW ITEM		NEW ITEM	NEW ITEM			IN PLACE		NEW ITEM
DOCKING		NEW ITEM		IN PLACE	IN PLACE						IN WORK				IN WORK
STATION KEEPING	NEW ITEM	NEW ITEM		NEW ITEM	NEW ITEM		NEW ITEM		NEW ITEM	NEW ITEM			IN PLACE	IN PLACE	NEW ITEM
ATTITUDE CONTROL	IN WORK	NEW ITEM	IN WORK	IN WORK	NEW ITEM	IN WORK	NEW ITEM		NEW ITEM	NEW ITEM			IN PLACE	IN WORK	IN WORK
POINTING CONTROL	NEW ITEM	NEW ITEM	IN WORK	IN WORK	NEW ITEM	IN WORK	NEW ITEM		NEW ITEM	NEW ITEM			IN PLACE	IN WORK	IN WORK
SHAPE CONTROL	NEW ITEM	NEW ITEM	NEW ITEM	NEW ITEM	IN WORK	NEW ITEM	NEW ITEM		NEW ITEM	NEW ITEM			NEW ITEM	NEW ITEM	NEW ITEM

IN PLACE → CURRENT TECHNOLOGY ADEQUATE
IN WORK → TECHNOLOGY DEVELOPMENT UNDERWAY
NEW ITEM → TECHNOLOGY DEVELOPMENT REQUIRED
(blank box) → NOT APPLICABLE

Figure 13

System Dynamics and Simulation of LSS

Simulations in Support of LSS

At this point in time simulations of LSS systems should begin with simplified simulation models. The models should grow in concert with the developments of LSS configurations using results from simulations and experience with equipment as complements. The past experiences with a rather complete simulation for the Sky Lab - gyros, environments, etc. allowed real time predictions in support of Flight events. The ability to keep an accurate budget of the propellant, to modify a maneuver as a result of a misdock, all contributed to the achievement of the long term mission. A similar goal seems appropriate for a LSS system with the effort beginning as relatively simple simulations to aid toward defining the most appropriate control concepts for Large Space Systems.

Applicability of Thermoplastic Composites For Space Structures

J.T. Hoggatt & M. Kushner
Boeing Aerospace Company
Seattle, Washington

Presented At
Large Space Systems Technology Seminar
Langley Research Center, Virginia

January 17-19, 1978

TEXT

APPLICABILITY OF THERMOPLASTIC COMPOSITES FOR SPACE STRUCTURES

INTRODUCTION (Figure 1)

THERMOPLASTIC COMPOSITES FOR USE IN THE DESIGN AND MANUFACTURE OF AEROSPACE HARDWARE ARE A RELATIVELY NEW DEVELOPMENT IN COMPOSITE TECHNOLOGY. THERMOPLASTIC COMPOSITES OFFER THE DESIGNER ANOTHER DIMENSION IN DESIGN ALTERNATIVES PHASED TOWARD A COST-EFFECTIVE, DESIGN-TO-COST APPROACH TO THE USE OF HIGH STRENGTH, HIGH MODULUS, LIGHTWEIGHT COMPOSITES FOR THE ACQUISITION OF AEROSPACE HARDWARE. THE BOEING COMPANY HAS BEEN A PIONEER IN THIS FIELD AND HAS SUCCESSFULLY DESIGNED, FABRICATED, AND ENVIRONMENTALLY TESTED THERMOPLASTIC COMPOSITE MAJOR HARDWARE. THE DISCUSSION, HEREIN, DEFINES A THERMOPLASTIC RESIN AND COMPARES THE STRUCTURAL AND ENVIRONMENTAL PROPERTIES AND THE FABRICATION AND REPAIRABILITY OF THE THERMOPLASTIC COMPOSITE WITH A TYPICAL EPOXY COMPOSITE. LOW LABOR COSTS EXHIBITED BY THE THERMOPLASTIC COMPOSITES MAKE THEM A PRIORITY CONSIDERATION FOR USE IN SPACE STRUCTURE.

LARGE SPACE SYSTEMS
TECHNOLOGY
BOEING AEROSPACE COMPANY

Thermoplastic Composites
Introduction

- COMPARISON OF THERMOPLASTIC vs THERMOSETTING RESINS AND COMPOSITES
- WHY THERMOPLASTIC COMPOSITES FOR SPACE APPLICATIONS
  - MANUFACTURING METHODS
  - REPAIRABILITY
  - COST SAVINGS
  - STRUCTURAL PROPERTIES
  - ENVIRONMENTAL STABILITY

Figure 1

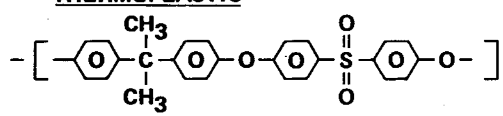
TEXT — THERMOPLASTIC vs. THERMOSETTING COMPOSITE (Figure 2)

THE TWO FORMULATIONS ARE GENERAL FOR A POLYSULFONE AND AN EPOXY RESIN. THE POLYSULFONE, A THERMOPLASTIC, TYPICALLY UNDERGOES A REVERSIBLE VISCOUS MELT AT 350°F, WHEREAS THE EPOXY, A THERMOSET, MAY SOFTEN SLIGHTLY BUT WILL DECOMPOSE RATHER THAN APPRECIABLY SOFTEN WITH INCREASED TEMPERATURE. CURRENTLY, TOP TEMPERATURE FOR THERMOPLASTICS IS 300°F OPERATIONAL USE. AS WE WILL SEE LATER, ENVIRONMENTAL AND SOLVENT RESISTANCE IS GOOD FOR SPACE HARDWARE USE.

LARGE SPACE SYSTEMS TECHNOLOGY
BOEING AEROSPACE COMPANY

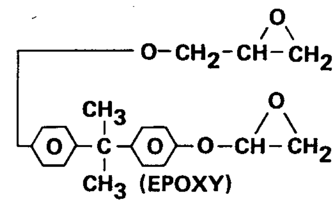
Thermoplastic vs Thermosetting Resin

THERMOPLASTIC



(POLYSULFONE)

THERMOSETTING



(EPOXY)

350^oF	SOFTENING TEMPERATURE (Tg)	DOES NOT SOFTEN
TO 300^oF	OPERATIONAL SERVICE TEMPERATURE	TO 350^oF
GOOD	ENVIRONMENTAL RESISTANCE	GOOD
GOOD EXCEPT FOR SELECTED SOLVENTS	SOLVENT RESISTANCE	GOOD
SINGLE COMPONENT	COMPOSITION	MULTIPLE COMPONENT

Figure 2

TEXT

GENERAL INFORMATION AND PREPREGS (Figure 3)

GLASS, KEVLAR,AND GRAPHITE CLOTH, TAPE OR UNIDIRECTIONAL FIBER REINFORCEMENTS ARE IMPREGNATED WITH THE THERMOPLASTIC RESIN BY SOLVENT OR HOT MELT TECHNIQUES. THEY ARE PURCHASED FROM THE MAJOR RESIN/IMPREGNATOR COMPANIES WHO SHIP USING STANDARD NONREFRIGERATED TECHNIQUES. THEY ARE THEN STORED AND HANDLED IN THE SHOP AT AMBIENT CONDITIONS UNTIL READY FOR USE.

FINISHED PARTS OR SUBASSEMBLIES ARE MACHINED AND FINISHED USING ESTABLISHED PROCEDURES.

LARGE SPACE SYSTEMS TECHNOLOGY
BOEING AEROSPACE COMPANY

Thermoplastic Composite General Information

IMPREGNATION TECHNIQUE	— SOLVENT OR HOT MELT
REINFORCEMENT AVAILABLE	— GLASS OR KEVLAR OR GRAPHITE CLOTH OR UNIDIRECTIONAL FIBERS
HOW PURCHASED	— RAW RESIN OR PREPREG REINFORCEMENT OR SHEET STOCK
HOW SHIPPED	— STANDARD PROCEDURES, NO REFRIGERATION REQUIRED
HOW STORED	— AMBIENT CONDITIONS, NO REFRIGERATION REQUIRED
HOW MACHINED	— STANDARD GRAPHITE COMPOSITE PROCEDURES
HOW PAINTED OR TOPCOATED	— STANDARD PAINTING PROCEDURES

Figure 3

TEXT

THERMOPLASTIC PREPREGS (Figure 4)

LIST OF AVAILABLE THERMOPLASTIC PREPREGS

Thermoplastic Prepregs

- HERCULES 3004 A-S 3-IN-WIDE TAPE
- DUPONT A-S/P1700 POLYSULFONE 6-IN-WIDE TAPE
- HEXCEL/T300 181 GRAPHITE FABRIC/P1700
- FIBERITE/T300 181 GRAPHITE FABRIC/P1700
- U.S. POLYMERIC T300/P1700 POLYSULFONE 12-IN-WIDE TAPE

Figure 4

TEXT

WHY THERMOPLASTIC COMPOSITES FOR SPACE APPLICATIONS (Figure 5)

THE FOLLOWING CHARTS ATTEMPT TO PORTRAY BRIEFLY THE ADVANTAGES OF THERMOPLASTIC COMPOSITES OVER EPOXY COMPOSITES FOR SPACE HARDWARE APPLICATIONS. THEY WILL SHOW A POTENTIAL FOR LOWER COSTS, INCREASED DESIGN EFFICIENCY AND EASE OF REPAIR. THIS COMBINED WITH GOOD STRUCTURAL PROPERTIES AND ENVIRONMENTAL STABILITY SHOULD MAKE THERMOPLASTIC COMPOSITES DESIRABLE FOR SPACE USE.

Why Thermoplastic Composites For Space Applications

- LOWER COMPONENT COSTS
  - LOWER FABRICATION COSTS
  - LOWER QUALITY ASSURANCE COSTS
  - LOWER SCRAPPAGE RATES
- INCREASED DESIGN EFFICIENCY
- EASE OF REPAIR
- GOOD STRUCTURAL PROPERTIES
- GOOD ENVIRONMENTAL STABILITY
- SINGLE COMPONENT SYSTEM

Figure 5

TEXT

LOWER COMPONENT COSTS (Figure 6)

THERMOPLASTIC COMPOSITE MATERIALS NEED NO REFRIGERATION. THEY ARE STORED AND HANDLED AT NORMAL SHOP CONDITIONS. CUTTING, SHAPING, PLY POSITIONING, APPLICATION OF PRESSURE AND TEMPERATURE, POST-FORMING AND ASSEMBLY METHODS ARE MORE LIKE FABRICATING WITH METALS LIKE ALUMINUM SHEET THAN LIKE "WET" EPOXY PREPREGS.

Thermoplastic Composites Lower Component Costs

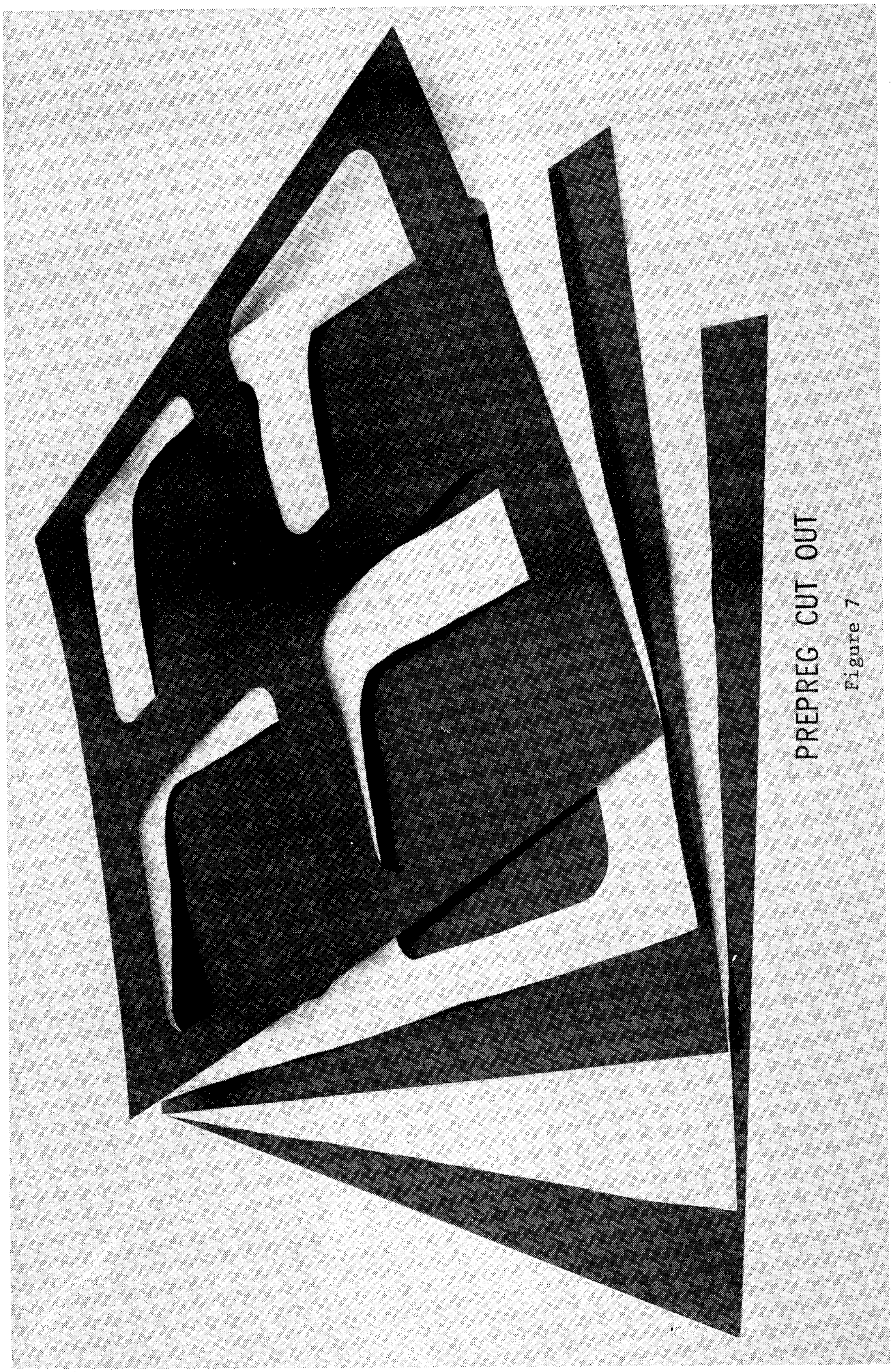
- STORAGE
  - AMBIENT CONDITION LIKE ALUMINUM SHEET
- LAMINATE CONSOLIDATION
  - CUTTING AND SHAPING
  - PLY POSITIONING
  - PRESSURE AND TEMPERATURE APPLICATION
- POST FORMING AND HANDLING
- ASSEMBLY METHODS

Figure 6

TEXT

PICTURE PREPREG CUT-OUT (Figure 7)

THIS PICTURE SHOWS HOW PLIES CAN BE CUT OUT AND LAYED UP IN PREPARATION TO FABRICATION OF A LAMINATE WITH DOUBLER OR REINFORCED AREAS. SUBSEQUENT APPLICATION OF PRESSURE (100 - 200 PSI) AND HEAT (425 - 650^{O}F) WILL FORM THE PART READY FOR THE FINISHING OPERATION.



PREPREG CUT OUT

Figure 7

TEXT

MANUFACTURING METHODS (Figure 8)

(FIGS. 8, 9 AND 10) LAMINATE CONSOLIDATION CAN BE ACCOMPLISHED USING TECHNIQUES CAPABLE OF APPLYING PRESSURE AND TEMPERATURE SIMULTANEOUSLY. FLAT SHEETS, FORMED PARTS, OR CONTINUOUS TAPE-LIKE ROLLS CAN BE FORMED. THESE PARTS CAN BE POST-FORMED BY HEATING AND PRESSING AGAIN AS MANY TIMES AS NECESSARY TO OBTAIN THE FINISHED SHAPE. (FIGS. 8, 11 AND 12) THE FINISHED SHAPE CAN BE MACHINED AS REQUIRED AND SUBSEQUENTLY BONDED OR MECHANICALLY ASSEMBLED INTO A SUBASSEMBLY. (FIGS. 8, 13 AND 14) PARTS SUCH AS TUBE FITTINGS CAN BE INJECTION MOLDED AND ASSEMBLED WITH THERMOPLASTIC COMPOSITE TUBING IN PRIME STRUCTURE SUCH AS SPACE METERING TRUSS.

Thermoplastic Composites Manufacturing Methods

Laminate consolidation:
- Roll-lamination
- Pultrusion
- Autoclave-lamination
- Press-lamination

Post-forming methods:
- Press (matched-die)
- Autoclave-molding
- Vacuum-forming
- Pultrusion

Bonding/joining methods:
- Fusion
- Adhesive-bonding
- Mechanical-fastening

Chopped fiber molding:
- Injection-molding
- Matched-die

Assembly methods:
- Fusion
- Adhesive-bonding

Figure 8

TEXT

PULTRUSION (Figure 9)

THIS IS A TYPICAL PULTRUSION MANUFACTURING FACILITY WHEREIN TAPE PASSES THROUGH A MICROWAVE PREHEAT CHAMBER, THEN TO A PRESSURE-TEMPERATURE COMPACTION CHAMBER AT 200 PSI AND 600^{o}F.

Thermoplastic Composite Pultrusion Facility

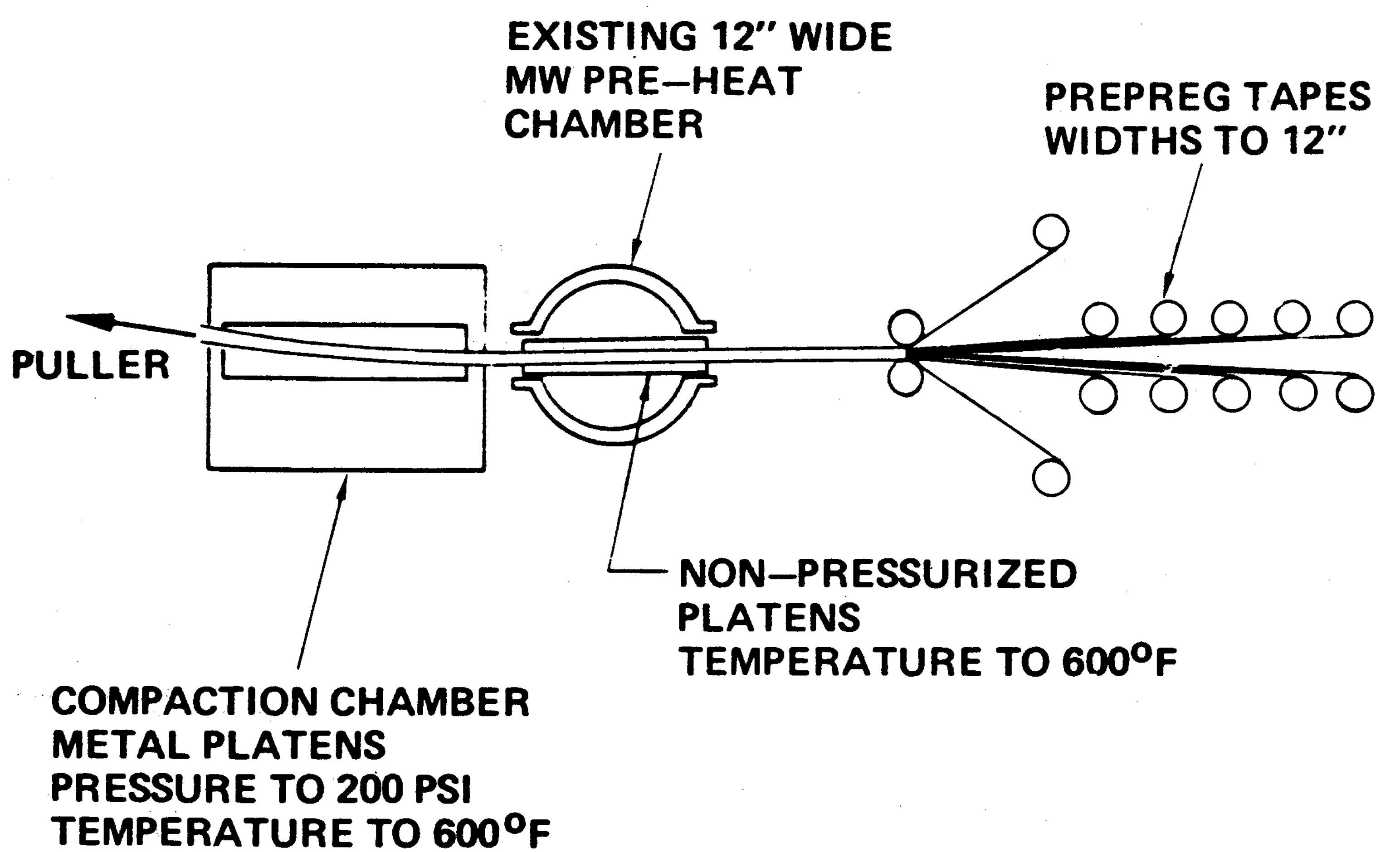


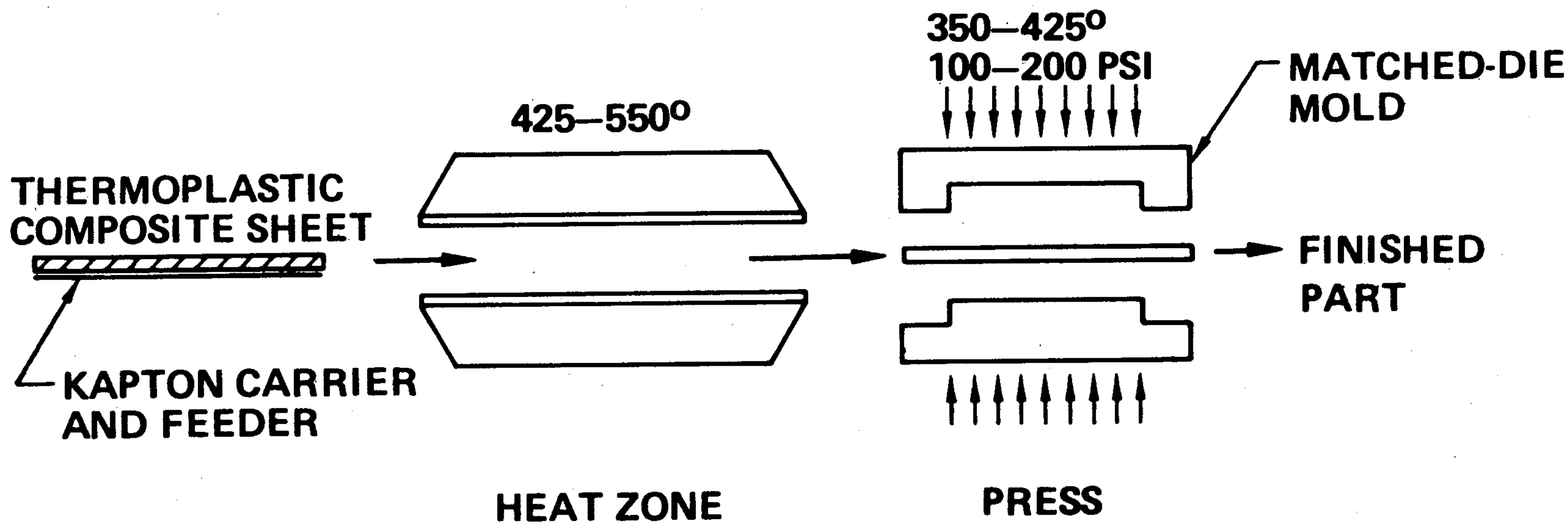
Figure 9

TEXT

PRESS FORMING (Figure 10)

STANDARD METHODS OF APPLICATION OF PRESSURE AND HEAT CAN BE USED FOR LAMINATE CONSOLIDATION. A PREHEAT ZONE IS USUALLY USED TO SOFTEN THE THERMOPLASTIC SO THAT IT WILL DRAPE WELL ON THE FINISHED MOLD WHEN FINISHED SHAPED PARTS ARE TO BE MADE AT THE SAME TIME LAMINATE CONSOLIDATION IS ACCOMPLISHED. THE PART CAN ALSO BE CONSOLIDATED IN A SHEET FORM, AND POST-FORMED BY A REPEAT OF THIS PRESS PROCESS.

Press Forming of Thermoplastic Composites



TOTAL ELAPSED PROCESSING TIME = 6–12 MINUTES

Figure 10

TEXT

PICTURE - POST-FORMED GRTP SHEETS (Figure 11)

THESE ARE TYPICAL PARTS WHICH HAVE BEEN POST-FORMED BY HEATING IN A VACUUM USING AN AIR ASSIST TO APPLY PRESSURE. THE PARTS WERE FORMED ON THE METAL TEMPLATE SHOWN.

Vacuum Formed Shear Web $[0^o/90^o]_s$ Sheet Stock

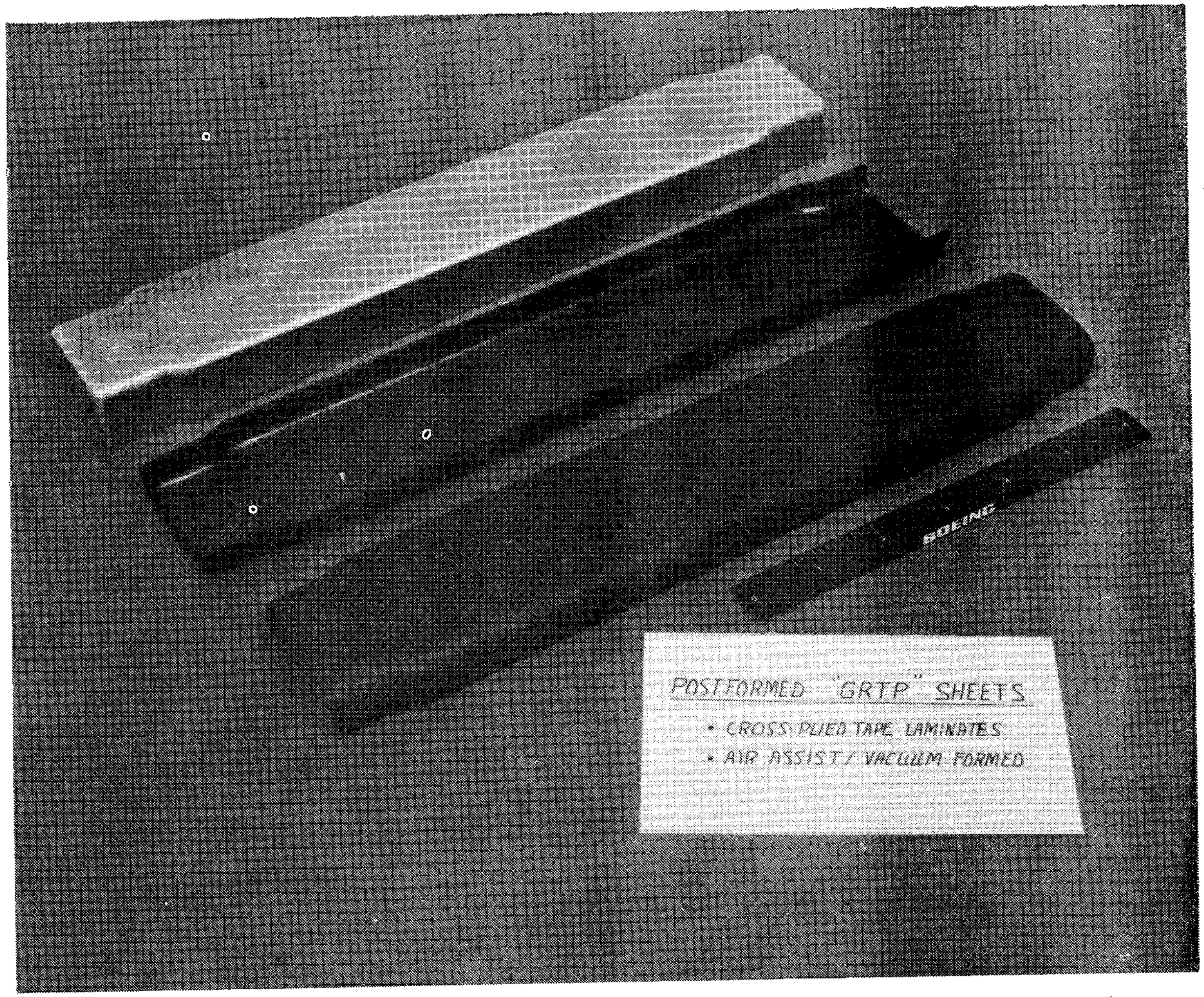
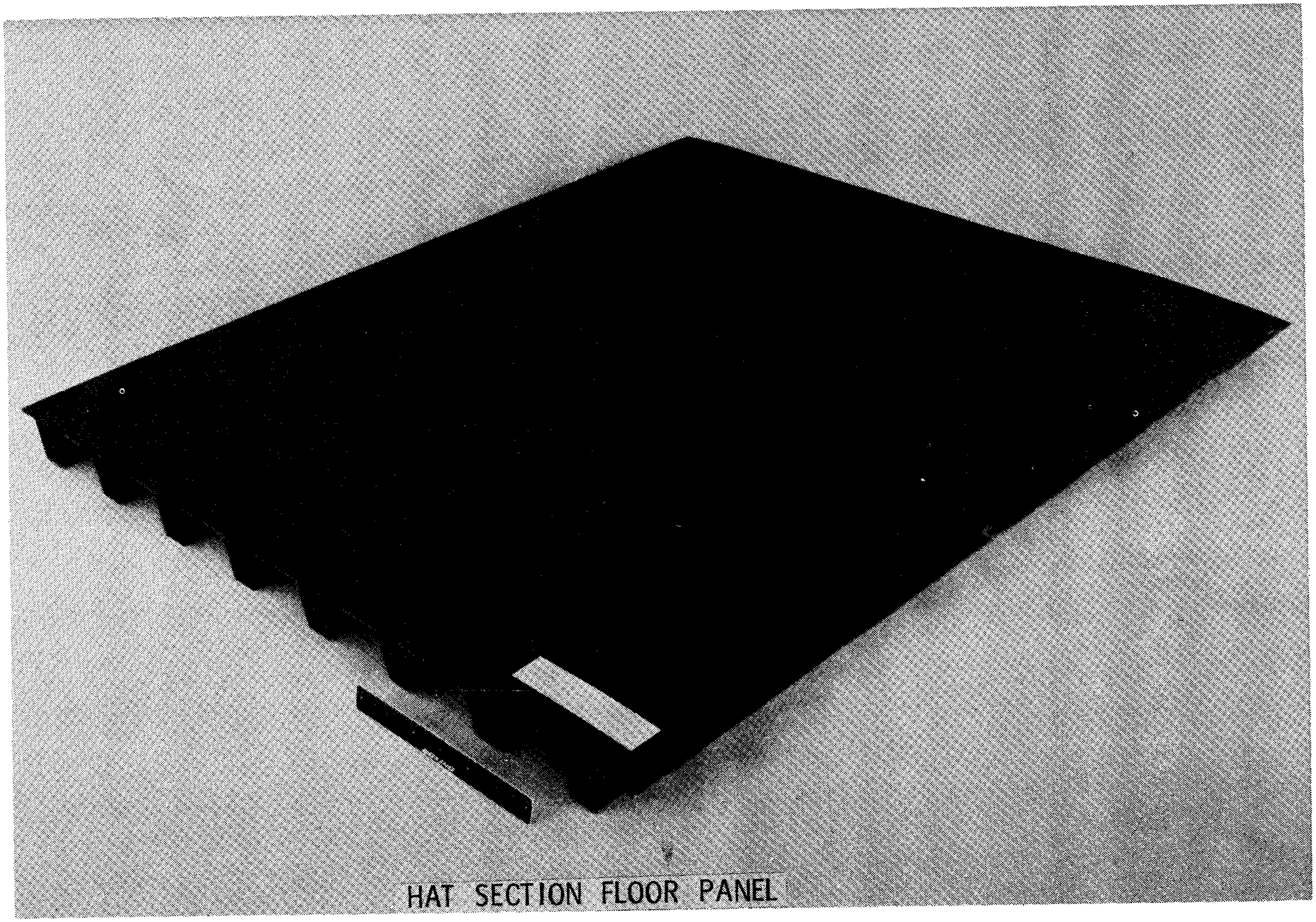


Figure 11

TEXT

PICTURE - HAT SECTION ASSEMBLY (Figure 12)

THIS IS A PICTURE OF AN ASSEMBLY COMPOSED OF A FLAT GRAPHITE THERMOPLASTIC COMPOSITE SHEET BONDED TO GRAPHITE THERMOPLASTIC COMPOSITE CORRUGATED PANEL FOR RIGIDITY. THE FACE SHEET AND THE CORRUGATED PANEL WERE INTEGRALLY CONSOLIDATED AND FUSED TOGETHER IN ONE OPERATION. THE PROCESS ELIMINATED THE APPLICATION OF ADHESIVES AND THE SUBSEQUENT BONDING OPERATION. REMOVABLE CORES WERE USED TO FORM THE CORRUGATIONS.



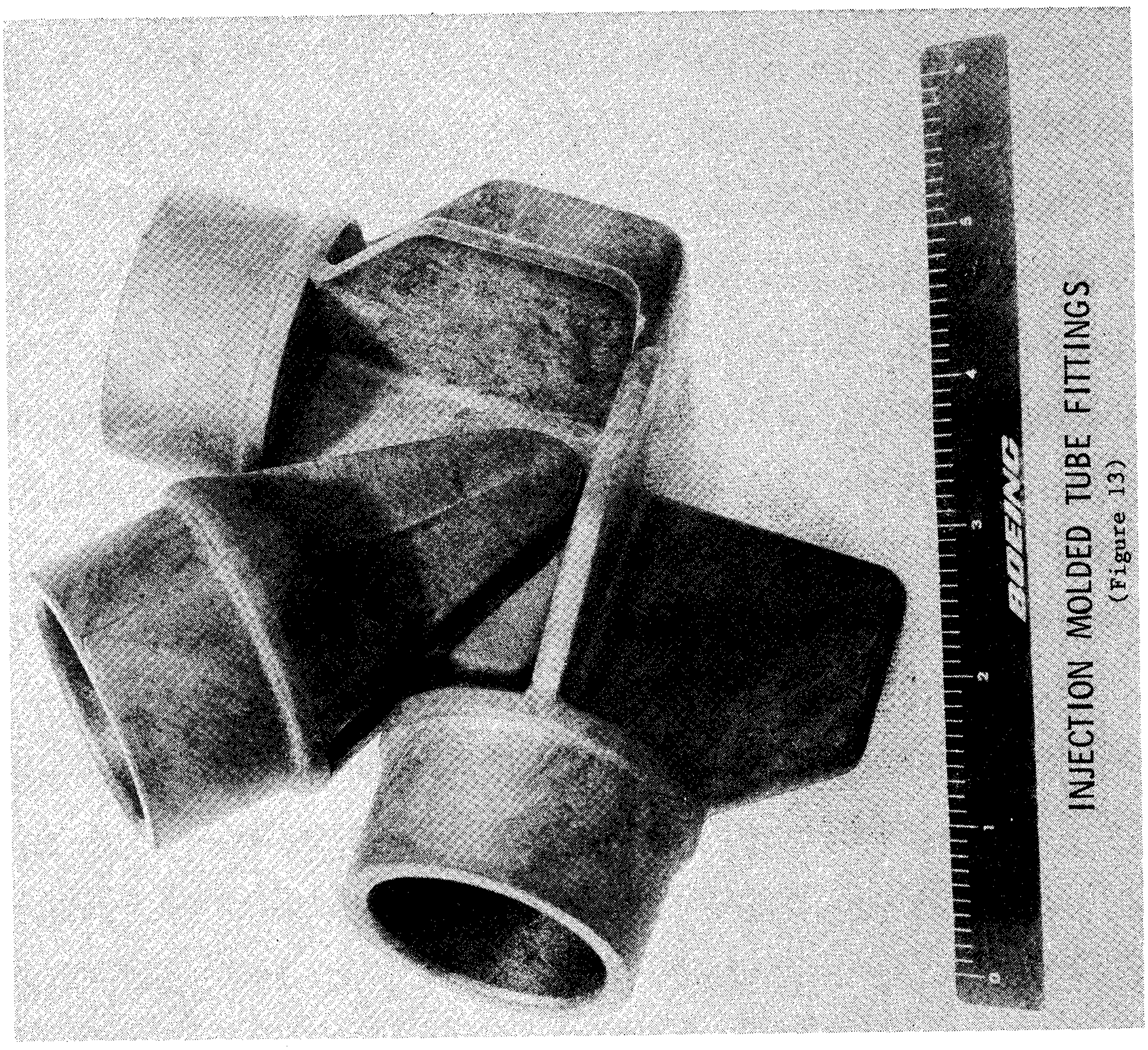
HAT SECTION FLOOR PANEL

Figure 12

TEXT

PICTURE - INJECTION MOLDED PARTS (Figure 13)

THIS IS A PICTURE OF GRAPHITE THERMOPLASTIC COMPOSITE MOLDED TUBE FITTINGS MANUFACTURED USING STANDARD INJECTION MOLDING TECHNIQUES FOR REINFORCED THERMO-PLASTICS.

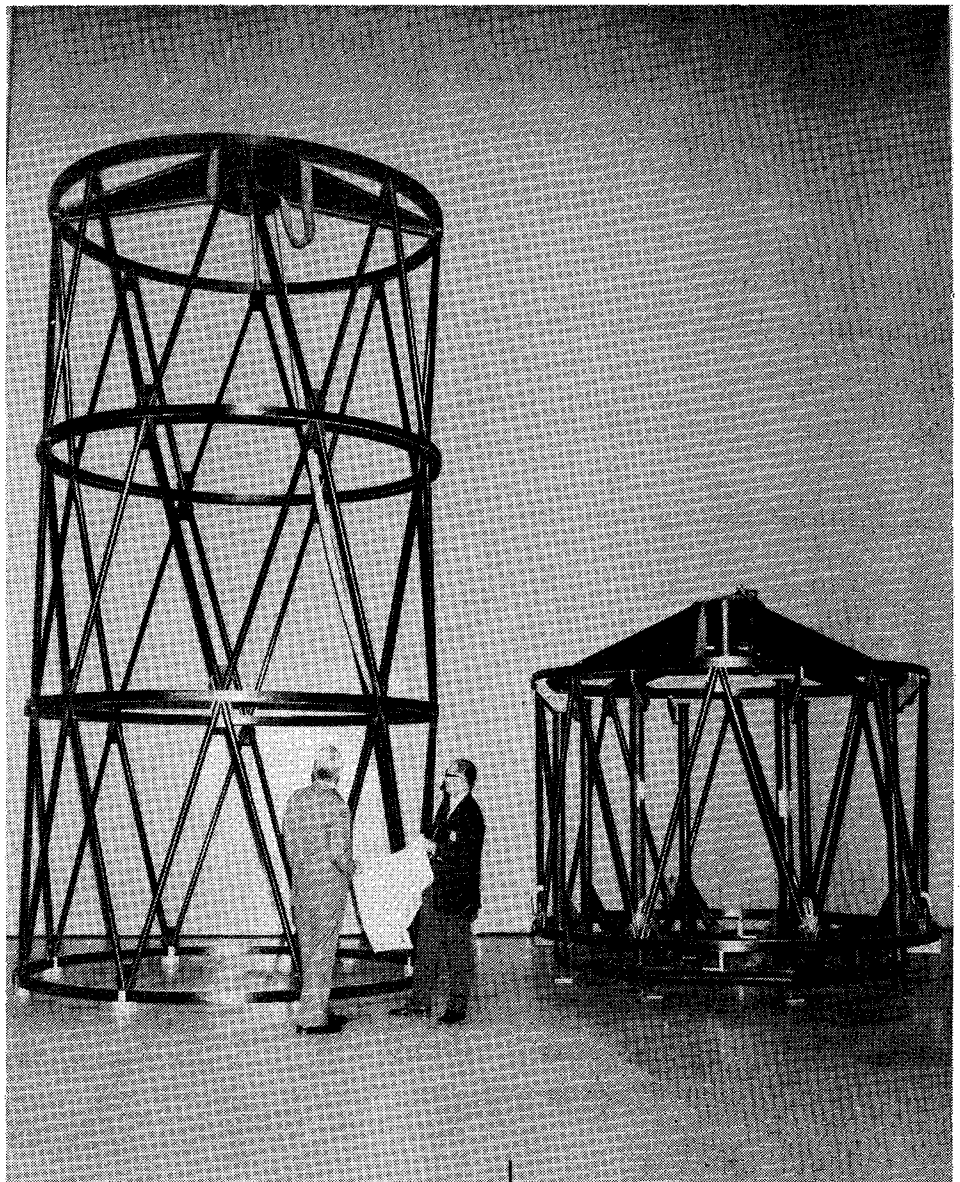


INJECTION MOLDED TUBE FITTINGS
(Figure 13)

TEXT

PICTURE - METERING TRUSS (Figure 14)

THIS PICTURE EXHIBITS TWO METERING TRUSSES. THE SHORTER OF THE TWO IS FABRICATED FROM GRAPHITE THERMOPLASTIC TUBES ASSEMBLED USING LAMINATED GRAPHITE FABRIC/ POLYSULFONE COMPOSITE FITTINGS. THE LARGER TRUSS IS AN EPOXY STRUCTURE. THE TWO STRUCTURES WERE FABRICATED PER THE SAME DESIGN.



GRAPHITE / PLASTIC METERING TRUSSES

Figure 14

TEXT | METHODS OF REPAIR (Figure 15)

(FIGS. 15, 16 AND 17) DAMAGED PARTS CAN BE RATHER EASILY REPAIRED BY REMOVING THE DAMAGE AND APPLYING REPLACEMENT MATERIAL BY ADHESIVE BONDING OR BY FUSING THE THERMOPLASTIC RESIN BY RADIANT, CONVECTIVE OR RESISTANCE HEATING OR BY ELECTRO-MAGNETIC BONDING. TENSILE TESTS OF REPAIRED SPECIMENS INDICATE SATISFACTORY STRUCTURAL PROPERTIES.

LARGE SPACE SYSTEMS
TECHNOLOGY
BOEING AEROSPACE COMPANY

Methods Of Repair

- ADHESIVE BONDING
- FUSION
  - RADIANT HEATING
  - CONVECTION HEATING
  - ELECTROMAGNETIC BONDING
  - RESISTANCE HEATING

Figure 15

TEXT — PICTURE - REPAIR METHOD (Figure 16)

THIS PICTURE SHOWS PATCHES USED AFTER DAMAGE HAS BEEN REMOVED AND HOLE PREPARED SO THAT A TAPERED PLUG CAN BE INSERTED PLY BY PLY. FINAL FUSION IS ACCOMPLISHED BY HEAT AND PRESSURE APPLICATION. ROOM-TEMPERATURE AND CONTACT PRESSURE BONDS HAVE ALSO BEEN MADE; HOWEVER, THE STRENGTH OF THE REPAIR IS NOT AS HIGH AS WITH HIGH TEMPERATURE - HIGH PRESSURE BONDING.

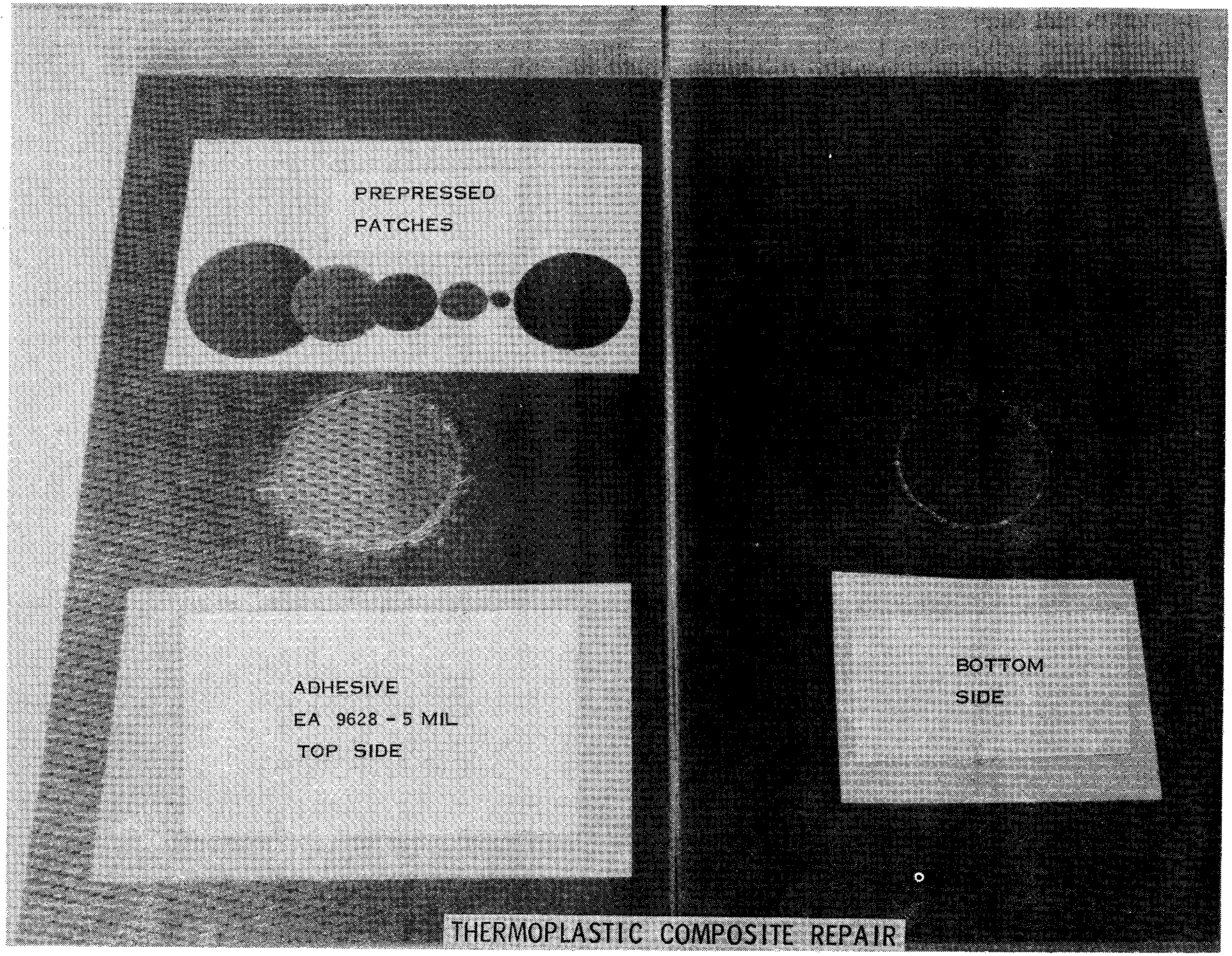
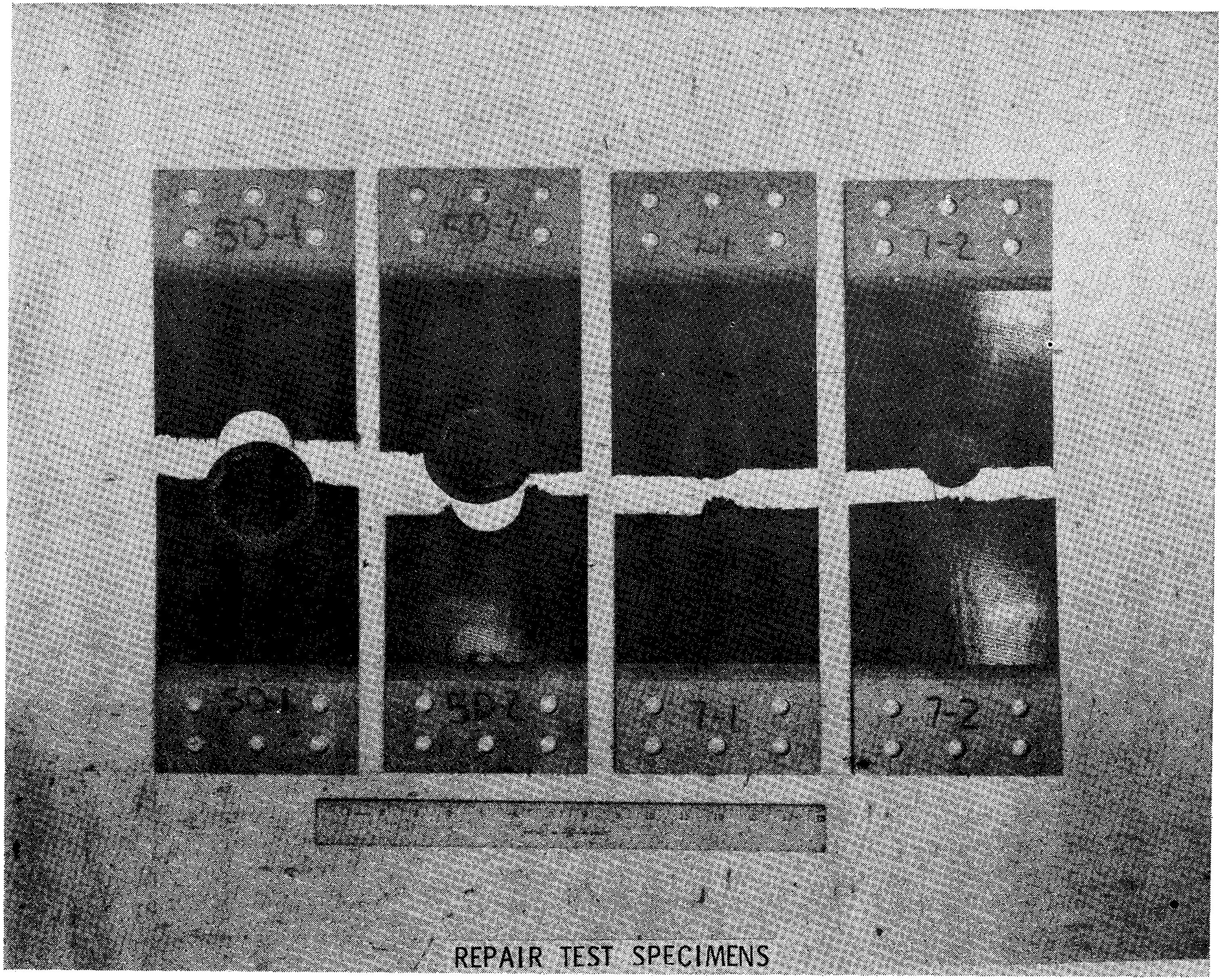

PREPRESSED
PATCHES
ADHESIVE
EA 9628 - 5 MIL
TOP SIDE
BOTTOM
SIDE
THERMOPLASTIC COMPOSITE REPAIR

Figure 16

TEXT PICTURE - REPAIR TEST SPECIMENS (Figure 17)

THESE ARE TYPICAL TEST SPECIMENS WITH VARYING SIZES OF REPAIRS AT THE CENTER OF THE SPECIMEN.



REPAIR TEST SPECIMENS

Figure 17

TEXT

REPAIR PANELS (Figure 18)

THE DATA HERE INDICATE THE EXCELLENCE OF ONE, TWO, AND THREE-HOLE FUSION AND BONDED REPAIRS. THEY COMPARE FAVORABLY WITH THE CONTROL (NO HOLE) SPECIMEN AND SUPERIOR TO THE SECOND CONTROL (ONE HOLE - NOT REPAIRED) SPECIMEN.

Repair Panels[a]

Panel	Area (in^2)	Load (lb)	Gross stress (psi)	Net stress (psi)
Control No. 1	0.1635	8,330[b]	50,948	50,948
Control No. 2 (1-in hole, no repair)	0.3276	8,830	26,593	32,343
1-in hole repair—fusion	↓	16,540	50,488	60,585
1-in hole repair—fusion	↓	17,100	52,197	62,636
2-in hole repair—fusion	↓	9,260	28,266	42,399
2-in hole repair—fusion	↓	9,600	29,304	43,956
3-in hole repair—fusion	↓	7,420	22,649	45,298
3-in hole repair—fusion	↓	6,720	20,512	41,024
1-in hole repair—bond	↓	17,250	52,655	63,186
1-in hole repair—bond	↓	16,220	49,511	59,413
1-in hole repair—fusion	↓	Fatigue 7.0×10^6 at 15,000 psi N.F., increase load		
1-in hole repair—fusion	↓	Fatigue 3.8×10^6 at 20,000 psi increased to 25,000 psi		

[a] Four plies of fabric at (0, 90)

[b] First test failed in grips at 66,000 psi, measured to 3 in width for second test.

Figure 18

TEXT

ELEMENT COST SAVING WITH GR/TP (Figure 19)

(FIGS.19 AND 20) COST SAVINGS IN LABOR ARE SIGNIFICANT WHEN COMPARING GRAPHITE REINFORCED EPOXY COMPOSITE LAYUP, CURE AND STANDARD MANUFACTURING TECHNIQUES TO SIMILAR OPERATIONS IN GRAPHITE-REINFORCED THERMOPLASTIC COMPOSITES. THE POSITIONING OF DOUBLERS, LOCAL HARD POINTS, AND THE FABRICATION OF NEAR NET DIMENSION PARTS ARE MUCH EASIER IN THERMOPLASTICS THAN IN EPOXY COMPOSITES.

THE ELEMENT COST SAVING IS CARRIED OVER INTO THE ASSEMBLY AS A SIGNIFICANT LABOR COST SAVING. HOWEVER, TO OVERCOME THE INCREASED MATERIAL AND TOOLING COSTS, OVERALL COST SAVINGS CAN BE APPRECIATED ONLY AFTER MULTIPLE ASSEMBLIES ARE PRODUCED.

LARGE SPACE SYSTEMS TECHNOLOGY
BOEING AEROSPACE COMPANY

Element Cost Savings With Gr/T.P.

			Savings*
Elements	T–Stiffeners		60%
	Hat–Stiffeners		65%
	L–Channel		72%
	Honeycomb Panels		30%
	Flat Panels		42%
	Complex Shapes	(Small)	70%
		(Large)	18%
Operations	Bonding		0
	Drilling		59%

*Over Gr/E (Labor)

Figure 19

TEXT

EFFECT OF LABOR SAVING ON COST (Figure 20)

THE GRAPHITE THERMOPLASTIC ASSEMBLY, AS TYPIFIED BY A STUDY OF THE COMPASS COPE HORIZONTAL STABILIZER, LABOR COSTS ARE LESS THAN THAT OF THE EPOXY GLASS ASSEMBLY. HOWEVER, DUE TO THE HIGHER COSTS OF MATERIALS AND TOOLING, MULTIPLE UNIT PRODUCTION MUST BE ACCOMPLISHED TO REALIZE A TOTAL COST SAVING. IT'S WORTH NOTING THAT EVEN WITH THE SIGNIFICANTLY HIGHER TOOLING COSTS THE THERMOPLASTIC COMPONENT BECOMES COST EFFECTIVE BY THE 10^{TH} UNIT.

Effect Of Labor Saving On Cost Of Complete Assembly

Compass Cope Horizontal Stabilizer Cost Estimate - Glass/Epoxy and Graphite/Polysulfone

ADVANCED COMPOSITE MATERIAL	NO. OF UNITS	PRODUCTION HOURS	MATERIAL DOLLARS	TOOLING HOURS	TOTAL COST DOLLARS
GLASS/EPOXY ($12/LB.)	1	1,068	600	810	56,940
	10	7,600	6,000	810	258,296
	100	46,730	60,000	810	1,486,178
GRAPHITE/POLYSULFONE ($65/LB.)	1	775	2,760	1,690	76,710
	10	5,515	27,600	1,690	243,747
	100	33,909	276,000	1,690	1,343,980

Figure 20

TEXT

P-1700/GRAPHITE FABRIC LAMINATE STRUCTURAL PROPERTIES (Figure 21)

(FIGS.21, 22 AND 23) GRAPHITE REINFORCED THERMOPLASTIC COMPOSITES EXHIBIT SIGNIFICANT TENSILE STRENGTH AND TENSILE MODULUS FROM -50^{o} TO 300^{o}F.

(FIGS.21, 24 AND 25) COMPRESSION AND FLEXURAL STRENGTH DECREASES WITH TEMPERATURE TO 300^{o}F; HOWEVER, THE MODULUS REMAINS FAIRLY CONSISTENT.

(FIGS.21 AND 26) INTERLAMINAR SHEAR DECREASES WITH INCREASE IN TEMPERATURE TO 300^{o}F.

IN GENERAL, STRUCTURAL PROPERTIES ARE KNOWN AND AVAILABLE FOR DESIGN UTILIZATION.

P-1700/Graphite Fabric Laminate Structural Properties

PROPERTY	TEST TEMPERATURE		
	+70 °F	+180 °F	+300 °F
TENSION			
STRENGTH, PSI	77,000	–	61,400
MODULUS, 10^6 PSI	10.0	–	10.4
COMPRESSION			
STRENGTH, PSI	56,000	–	36,000
MODULUS, 10^6 PSI	9.2	–	10.6
FLEXURAL (0°)			
STRENGTH, PSI	107,000	92,000	69,000
MODULUS, 10^6 PSI	7.7	7.1	7.2
FLEXURAL (90°)			
STRENGTH, PSI	97,000	80,000	66,000
MODULUS, 10^6 PSI	7.7	6.8	6.7
INTERLAMINAR SHEAR			
STRENGTH, PSI	8,740	–	4,390

Figure 21

TEXT

TEMPERATURE vs. TENSILE STRENGTH (Figure 22)

THE TENSILE STRENGTH VARIES ONLY SLIGHTLY FROM -50° TO 300°F. THIS IS ON A LAMINATE WHOSE FIBER VOLUME IS 60%.

Effect of Temperature on Tensile Strength

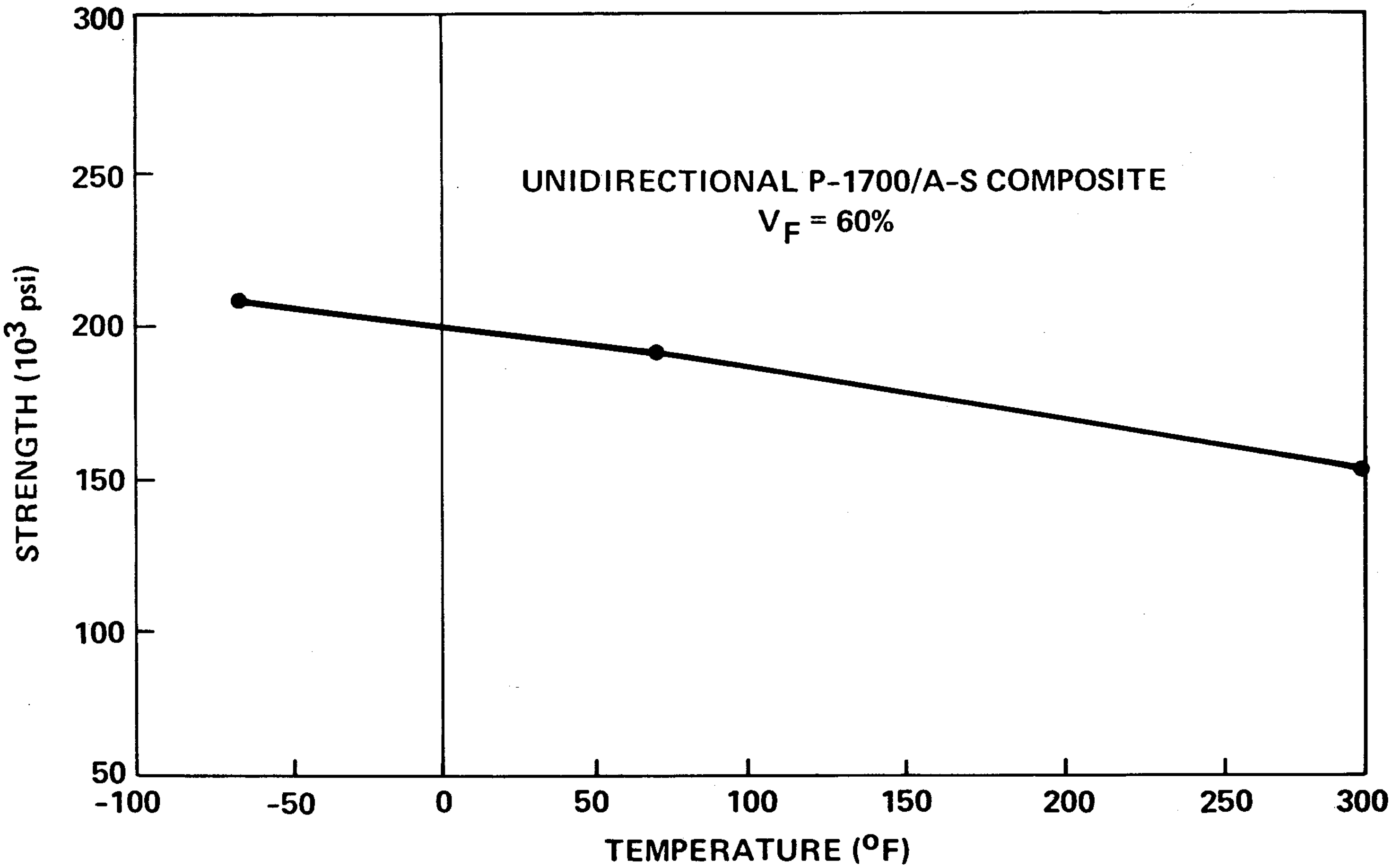


Figure 22

TEXT

TEMPERATURE vs. TENSILE MODULUS (Figure 23)

THE TENSILE MODULUS IS NOT APPRECIABLY AFFECTED BY TEMPERATURE EXPOSURES -50^{o} TO 300^{o}F. THIS IS ON A LAMINATE WHOSE FIBER VOLUME IS 60%.

Effect of Temperature on Tensile Modulus

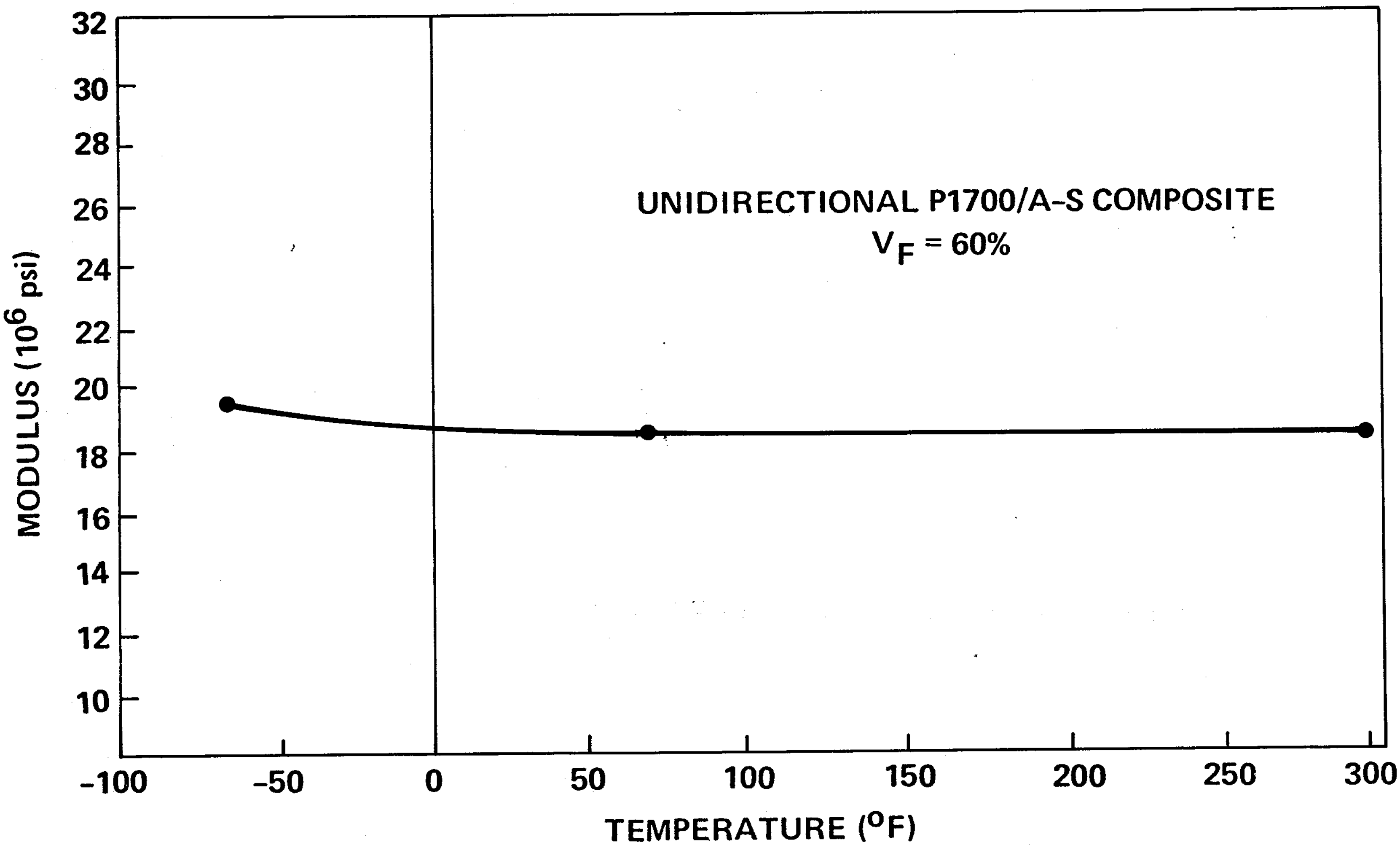


Figure 23

TEXT

TEMPERATURE vs. FLEXURAL STRENGTH (Figure 24)

FLEXURAL STRENGTH DECREASES WITH INCREASE IN TEMPERATURE FOR UNIDIRECTIONAL (0), BIDIRECTIONAL $(0 - 90)_S$, AND FABRIC LAMINATES. THIS IS TO BE EXPECTED SINCE THE RESIN SOFTENS WITH INCREASE IN TEMPERATURE.

Effect of Temperature on Flexure Strength— Graphite/ Polysulfone Composites

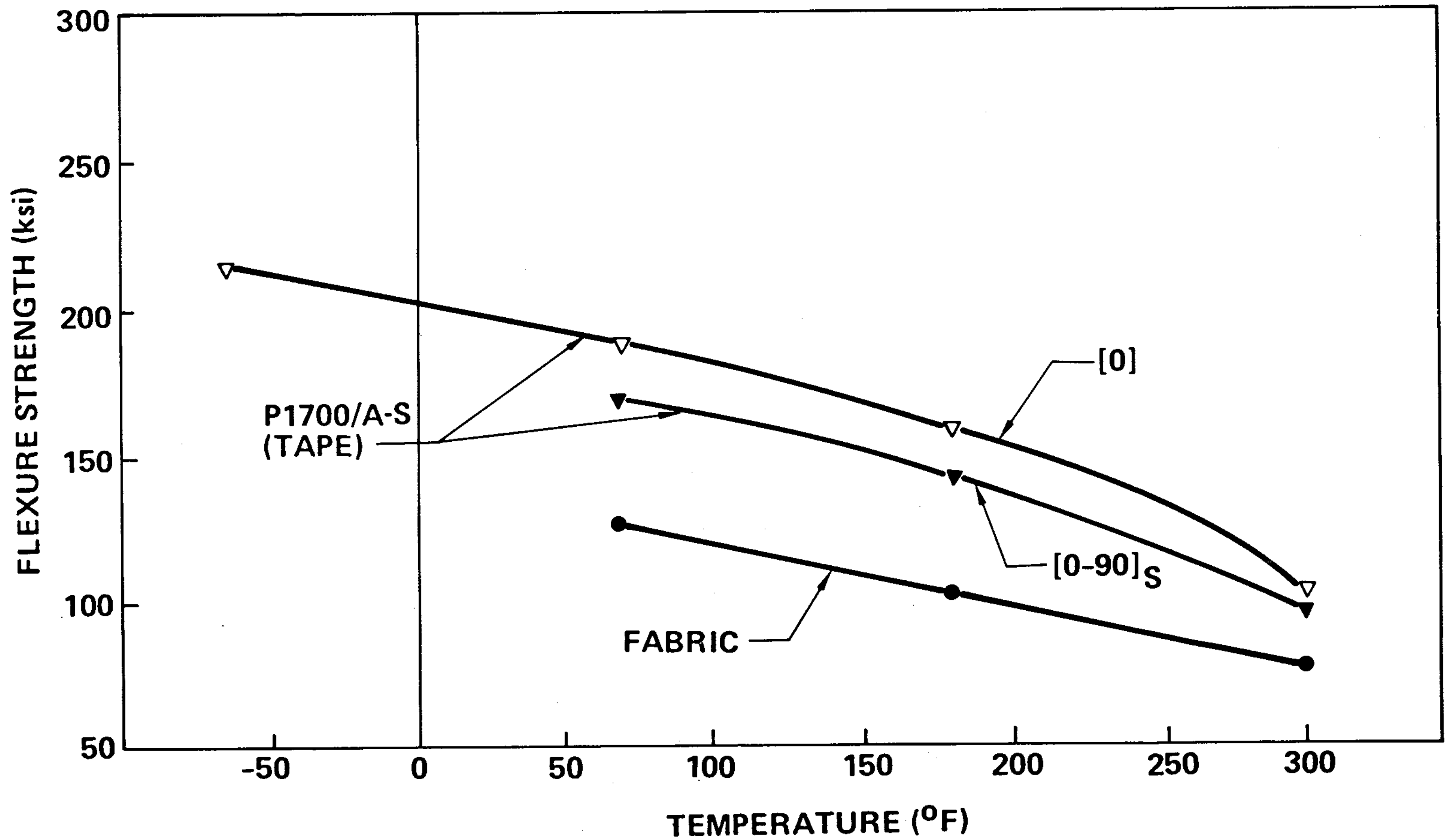


Figure 24

TEXT

TEMPERATURE vs. FLEXURE MODULUS (Figure 25)

THE FLEXURE MODULUS IS NOT AFFECTED BY RAISE IN TEMPERATURE FROM -50^{o} TO 300^{o}F. THIS IS TRUE OF UNIDIRECTIONAL (0), BIDIRECTIONAL $(0 - 90)_S$ AND FABRIC LAMINATES.

Effect Of Temperature On Flexure Modulus

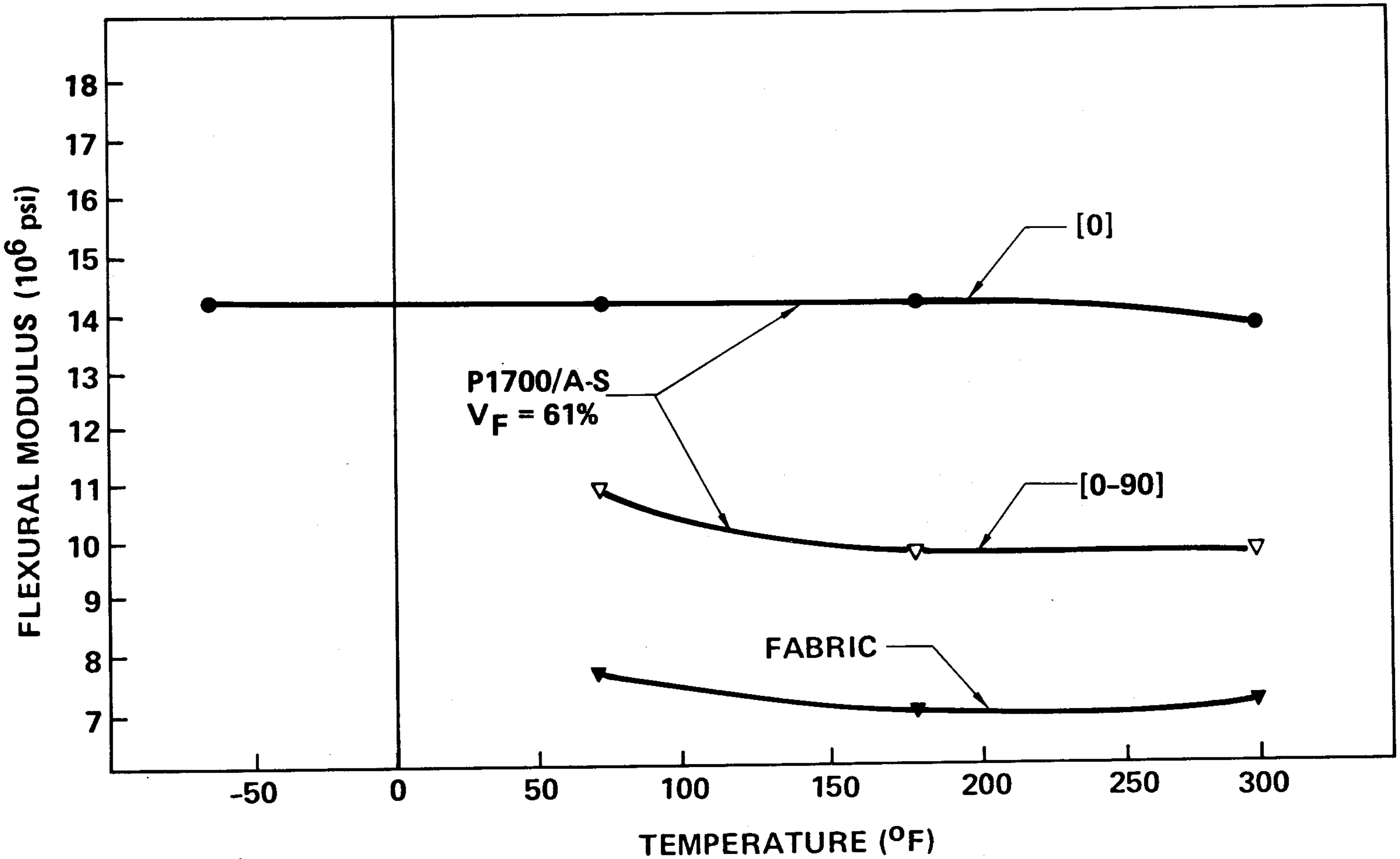


Figure 25

TEXT

TEMPERATURE vs. INTERLAMINAR SHEAR (Figure 26)

THE INTERLAMINAR SHEAR STRENGTH DECREASES WITH THE INCREASE IN TEMPERATURE TO 300°F. THIS LAMINATE HAD A FIBER VOLUME OF 58%.

Effect of Temperature on Interlaminar Shear

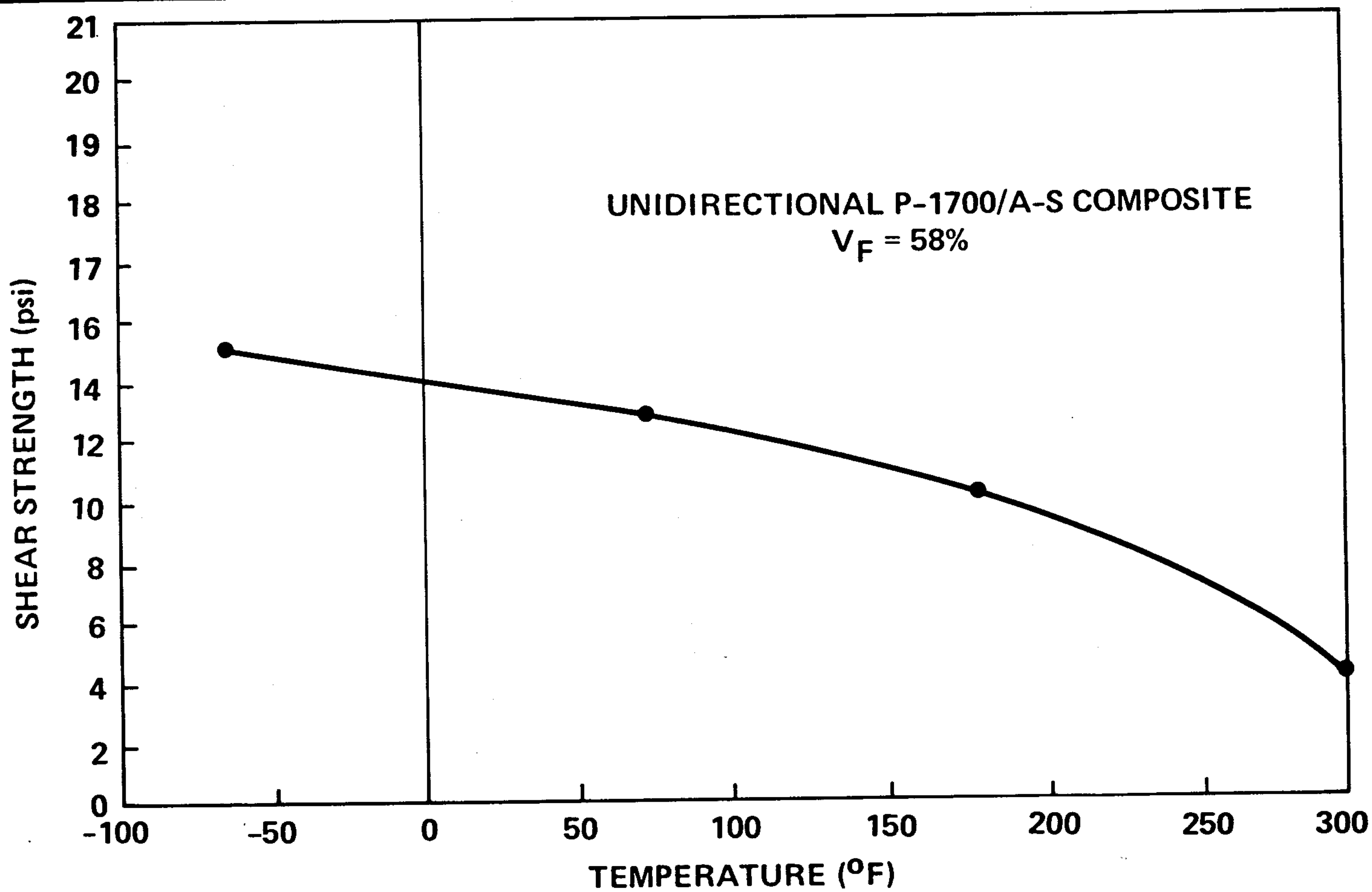


Figure 26

TEXT

ENVIRONMENTAL STABILITY (Figure 27)

THERMOPLASTIC COMPOSITES ARE ESSENTIALLY ENVIRONMENTALLY STABLE. THEY EXHIBIT LOW MOISTURE ABSORPTION, GOOD THERMAL STABILITY TO 300°F INCLUDING RESISTANCE TO MICRO-CRACKING WHEN EXPOSED TO SEVERE TEMPERATURE CYCLING. THEY OUTGAS IN A VACUUM ENVIRONMENT WITHIN ACCEPTABLE LIMITS OF 1% WEIGHT LOSS AND 0.1% VCM. THERE IS CURRENTLY, AS WITH EPOXY SYSTEMS, A LACK OF DATA ON RADIATION STABILITY; HOWEVER, BASED ON AVAILABLE THERMOPLASTIC RESIN RADIATION RESISTANCE DATA, IT IS SUSPECTED THAT THE COMPOSITE WILL SHOW SATISFACTORY RESISTANCE. CONTRARY TO POPULAR BELIEF, THERMOPLASTICS ARE ESSENTIALLY RESISTANT TO SOLVENT ATTACK EXCEPT FOR CERTAIN HALOGENATED SOLVENTS. THE COMPOSITES CAN BE PRIMED AND PAINTED USING STANDARD MIL-SPEC PROCEDURES AND WILL RESIST LONG TIME (28 DAY) CONTINUOUS EXPOSURE TO SUCH LIQUIDS AS JP-4 FUEL, HYDRAULIC OIL (MIL-H-5606), SYNTHETIC LUBRICANT (MIL-L-7807) AND POLYSULFIDE SEALANTS. SURFACE CLEANING OF THE THERMOPLASTIC COMPOSITES CAN BE ACCOMPLISHED WITH COMMON NON-HALOGENATED SOLVENTS (I.E., NAPHTHA, ALCOHOLS, ETC.)

Environmental Stability
Thermoplastic Composites

• MOISTURE ABSORPTION	LOW (.25% @ 75^{o}F)
• THERMAL STABILITY	GOOD TO 300^{o}F GOOD STABILITY CYCLING -320^{o}F TO +300^{o}F (MINIMAL MICROCRACKING)
• VACUUM OUTGASSING	TOTAL WEIGHT LOSS = 0.38% VACUUM CONDENSIBLES = 0.06%
• SPACE RADIATION STABILITY	UNKNOWN - HOWEVER,SOME DATA ARE AVAILABLE ON RESIN PROPERTIES
• SOLVENT STABILITY	GOOD EXCEPT FOR CHLORINATED SOLVENTS

Figure 27

TEXT SUMMARY (Figure 28)

TO SUMMARIZE, THERMOPLASTIC COMPOSITES OFFER THE SPACE SYSTEM DESIGNER A LOW-COST VERSATILE MATERIAL FOR SPACE SYSTEM HARDWARE DESIGN, CAPABLE OF BEING HANDLED, MACHINED AND ASSEMBLED IN THE SHOPS USING ESSENTIALLY STANDARD MANUFACTURING TOOLS AND PROCEDURES. THE FABRICATED STRUCTURE CAN BE READILY REPAIRED AND WILL EXHIBIT PREDICTABLE STRUCTURAL AND ENVIRONMENTAL STABILITY.

Thermoplastic Composites Summary

THERMOPLASTIC COMPOSITES FOR LARGE SPACE STRUCTURE WHEN COMPARED TO EPOXY COMPOSITES

- LOW COST
- GREATER DESIGN ALTERNATIVES
- MANUFACTURING AND ASSEMBLY VERSATILITY
- REPAIR CAPABILITY
- GOOD STRUCTURAL AND ENVIRONMENTAL STABILITY
- SINGLE COMPONENT SYSTEM

Figure 28

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Applicability of Thermoplastic Composites for Space Structures

Test Data from Materials Exposed to Space Environments

At the present time the limited understanding of the degradation modes - ultra violet as surface damage, charged particles as internal damage, has resulted in a correspondingly limited capability to perform accelerated testing on materials. Confirmatory testing from samples exposed on-orbit appear to be a real requirement in achieving such an understanding. The need exists to proceed immediately toward long term testing of material in actual orbital environments.

SPACECRAFT CHARGING AND PLASMA INTERACTION IMPLICATIONS FOR LARGE SPACE SYSTEMS

- E. MILLER
- M. STAUBER
- M. ROSSI
- W. FISCHBEIN

WHAT IS THE PROBLEM? (Figure 1)

Many conventional satellites have experienced performance anomalies and possible damage in geosynchronous orbit (GEO). The majority of anomalies correlate with environmental disturbances (magnetic substorms) when the spacecraft encounters high energy electrons which differentially charge exposed surfaces.

Although a spacecraft in low earth orbit (LEO) encounters only low energy plasma, high voltage systems will accelerate plasma particles, resulting in surface charging and damage possibly more severe than in GEO. The high particle density in LEO could also contribute to high power losses for multikilovolt systems.

Spacecraft charging studies have been conducted since the early 1970's. Little is known, however, about specific discharge mechanisms, plasma interactions, and scale effects associated with very large spacecraft. The large area, low density character, and extensive use of non-conducting materials could have a major impact on the performance and survivability of many Large Space Systems (LSS).

WHAT IS THE PROBLEM?

- "DISTURBED" GEO ENVIRONMENT MAY PRODUCE HIGH DIFFERENTIAL CHARGING & ELECTRICAL DISCHARGES RESULTING IN:
  - MATERIAL/COMPONENT DEGRADATION
  - EQUIPMENT MALFUNCTION/DAMAGE
- HIGH VOLTAGE (MULTIKILOVOLT) SYSTEMS IN LEO MAY:
  - INDUCE HIGH DIFFERENTIAL CHARGING/ELECTRICAL DISCHARGES, SPUTTERING & RADIATION DAMAGE
  - EXPERIENCE LARGE POWER LOSS THRU PLASMA
- SPECIAL LSS CHARACTERISTICS COULD COMPOUND CHARGING/PLASMA INTERACTION PROBLEMS
- SPACECRAFT CHARGING STUDIES TO DATE EMPHASIZE CONVENTIONAL DESIGNS – LITTLE EFFORT ON LSS

Figure 1

<u>MAGNETOSPHERE</u> (Figure 2)

The magnetosphere is the region of the earth's space in which the geomagnetic field exerts a dominant influence on the motion of low-energy plasma and fast-charged particles. The solar wind, primarily ionized hydrogen gas, distorts the earth's dipole magnetic field and forms the magnetopause boundary which excludes all impinging plasma particles. A detached shock wave is formed on the sun side of the magnetosphere at about 9 earth radii (R_e), and an elongated magnetotail is formed on the opposite side which extends for hundreds of earth radii.

The magnetosphere is immersed in the weak interplanetary magnetic field whose direction depends on the rotational position of the sun from which it emanates. This field combines with the dipole field to create open field lines allowing plasma exchange between the ionosphere and interplanetary space.

The plasma sheet, a broad ($\sim 4\ R_e$) region of dilute, warm ($n \sim 1\ cm^{-3}$, $T \sim 500$ ev) plasma lying in the equatorial plane, extends from the dawn to dusk side of the magnetotail. During quiet periods, a geosynchronous satellite at $6.6R_e$ will pass through the plasma sheet and other regions where particle energies are generally less than 10 ev with densities from 10 to 1000 cm^{-3}. During disturbed periods (magnetic substorms), the character of the plasma sheet changes rapidly in the region from 3 to 12 R_e where the quiescent plasma is replaced by an energetic plasma with electron energies up to 20 kev.

MAGNETOSPHERE (NOON-MIDNIGHT PLANE)

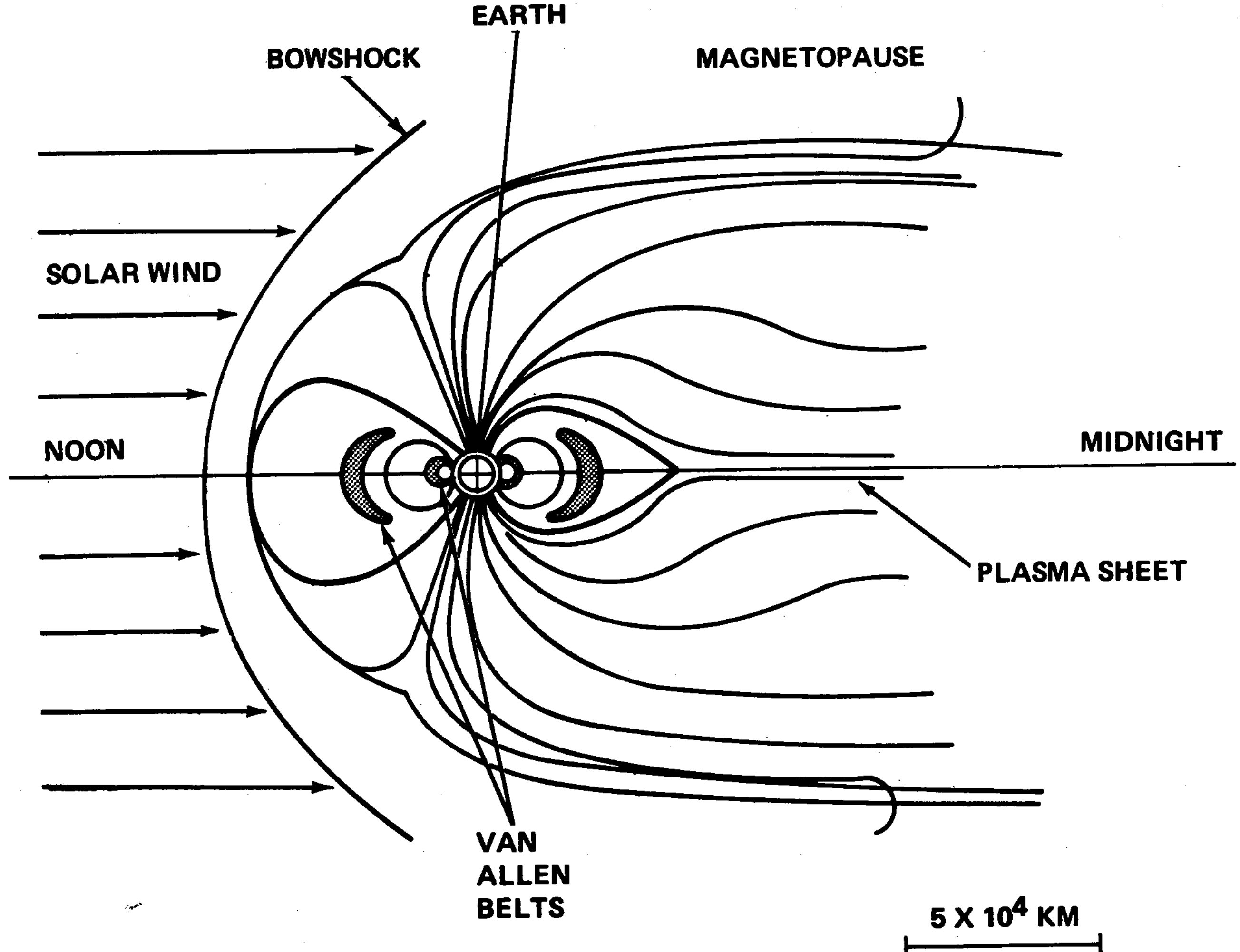


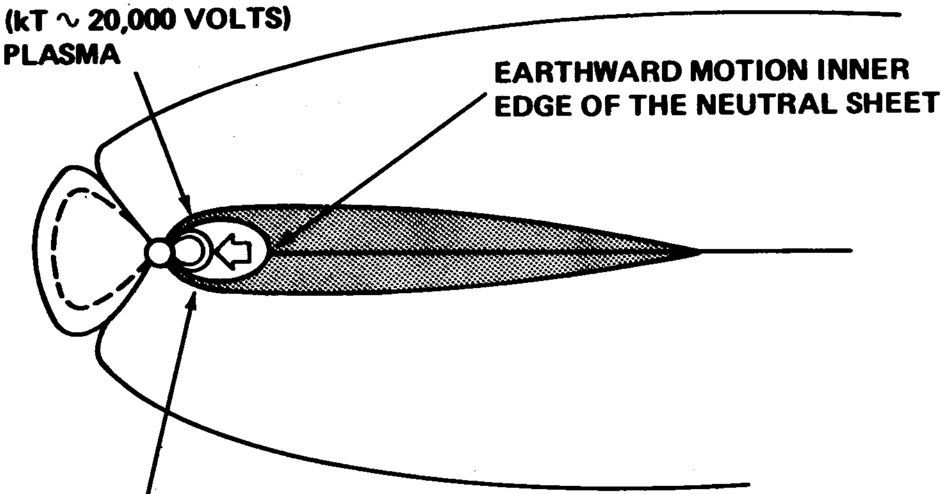
Figure 2

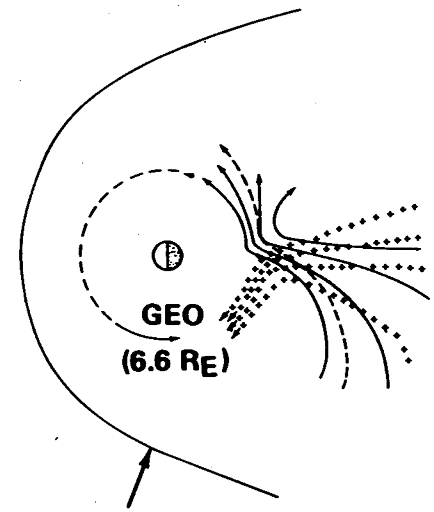
MAGNETIC SUBSTORMS (Figure 3)

Magnetic substorms occur as isolated events every few days or hours, or they may occur in rapid sequence. There is yet no conclusive theory to explain the generation and development of the substorm. It appears, however, that substorms occur when the interplanetary magnetic field, convected by the solar wind, develops a southward-directed component. The magnetosphere is compressed, magnetic field lines merge, and stored energy is somehow released resulting in a high energy plasma flow toward the earth. The energy of the plasma particles increases as they encounter stronger fields and changing field line curvatures and gradients. Magnetic gradient and curvature effects cause electrons and ions to drift into different hemispheres. The preferential drift of high energy electrons in the local midnight to dawn region is believed responsible for most high differential spacecraft charging and performance anomalies observed on geosynchronous satellites.

GEO and LEO plasma characteristics are compared in the table. LEO is characterized by very dense, low-energy electrons where, except for high voltage systems, spacecraft charging is not a problem. During magnetic substorms, the low energy/density electrons at GEO are replaced by a tenuous plasma containing high-energy electrons.

MAGNETIC STORMS





PLASMA CHARACTERISTICS (TYPICAL)

	LEO	GEO QUIESCENT	GEO SUBSTORM
AVG ELECTRON ENERGY (EV)	~ 1	1–500	5000–20,000
PLASMA DENSITY (PARTICLES/CM^3)	10^5–10^6	~ 100	<10

Figure 3

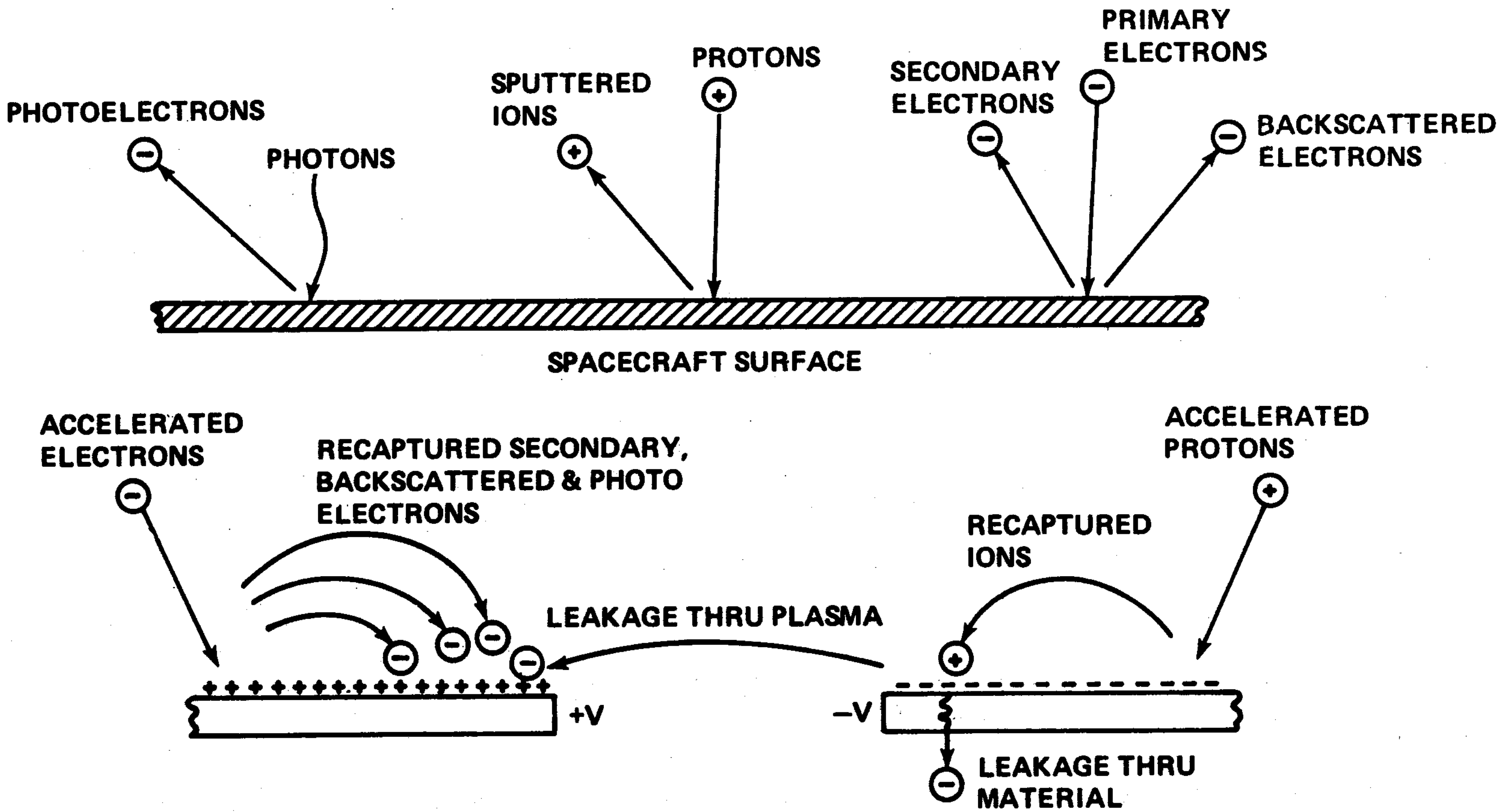
MAJOR SPACECRAFT CHARGING PROCESSES (Figure 4)

A variety of processes and properties determine the magnitude and polarity to which a surface will charge in a plasma. Primary electrons and sputtered ions contribute to negative charging, while photoelectrons, secondary and backscattered electrons which escape result in a positively-charged surface. If the surface is charged, it will attract particles of opposite polarity and recapture some portion of those particles which might otherwise escape. In addition, electrons may leak between differentially charged surfaces through the plasma or through the bulk material.

The charging processes continue until the algebraic sum of all currents equals zero. An electrically powered structure will float at some equilibrium voltage with respect to the plasma such that $J(+)A(+) = J(-)A(-)$, i.e., total negative and positive current-area products are equal.

Charge levels depend on spacecraft configuration, composition, orientation, and local plasma characteristics.

MAJOR SPACECRAFT CHARGING PROCESSES



CHARGE MAGNITUDE DEPENDENCE

- SOLAR INTENSITY
- LOCAL PARTICLE DENSITY/ENERGY
- SURFACE POTENTIAL
- MATERIAL PROPERTIES
  - SECONDARY/PHOTOEMISSION
  - RESISTIVITY
  - BREAKDOWN POTENTIAL

Figure 4

ELECTROSTATIC CHARGING/DISCHARGING (PASSIVE ELEMENTS) (Figure 5)

Several charge/discharge mechanisms may be responsible for spacecraft performance anomalies and material damage.

A charged dielectric will polarize and large electric field stresses can exist in a thin sheet of material. If the breakdown potential is exceeded, discharge will occur through the bulk. Besides possible damage to the bulk, some material might be removed from the surface; this contaminant could redeposit on nearby surfaces. If one surface is metallized, the contaminant could include vaporized metal.

Very high current metal to metal discharges are another possibility where ungrounded metallized surfaces are in close proximity to grounded structure. If the metallized layers of thermal blankets are electrically isolated, the layers will charge differently; and, high current discharges could occur between layers or from any layer to the conducting structure.

Other mechanisms which have been postulated include bilayer and Malter discharge. Bilayer discharge may occur immediately within the outer surface layer of a dielectric and arises from electrons that penetrate the bulk and the positive surface charge resulting from secondary emission. Insulating film deposited on a metal surface can form a lossy semiconductor junction which can break down at potentials of only a few tens of volts. These (Malter) discharges can vaporize the metal as well as damage the insulating surface. Bilayer and Malter discharges may be partly responsible for spacecraft anomalies which have been recorded in the absence of major substorms.

Studies have shown that when electrical discharge occurs in a dielectric, negatively charged material (electrons, surface atoms) is repelled to several centimeters above the surface. This acts as a dipole antenna producing large, directional r.f. noise pulses.

ELECTROSTATIC CHARGING/DISCHARGES (PASSIVE ELEMENTS)

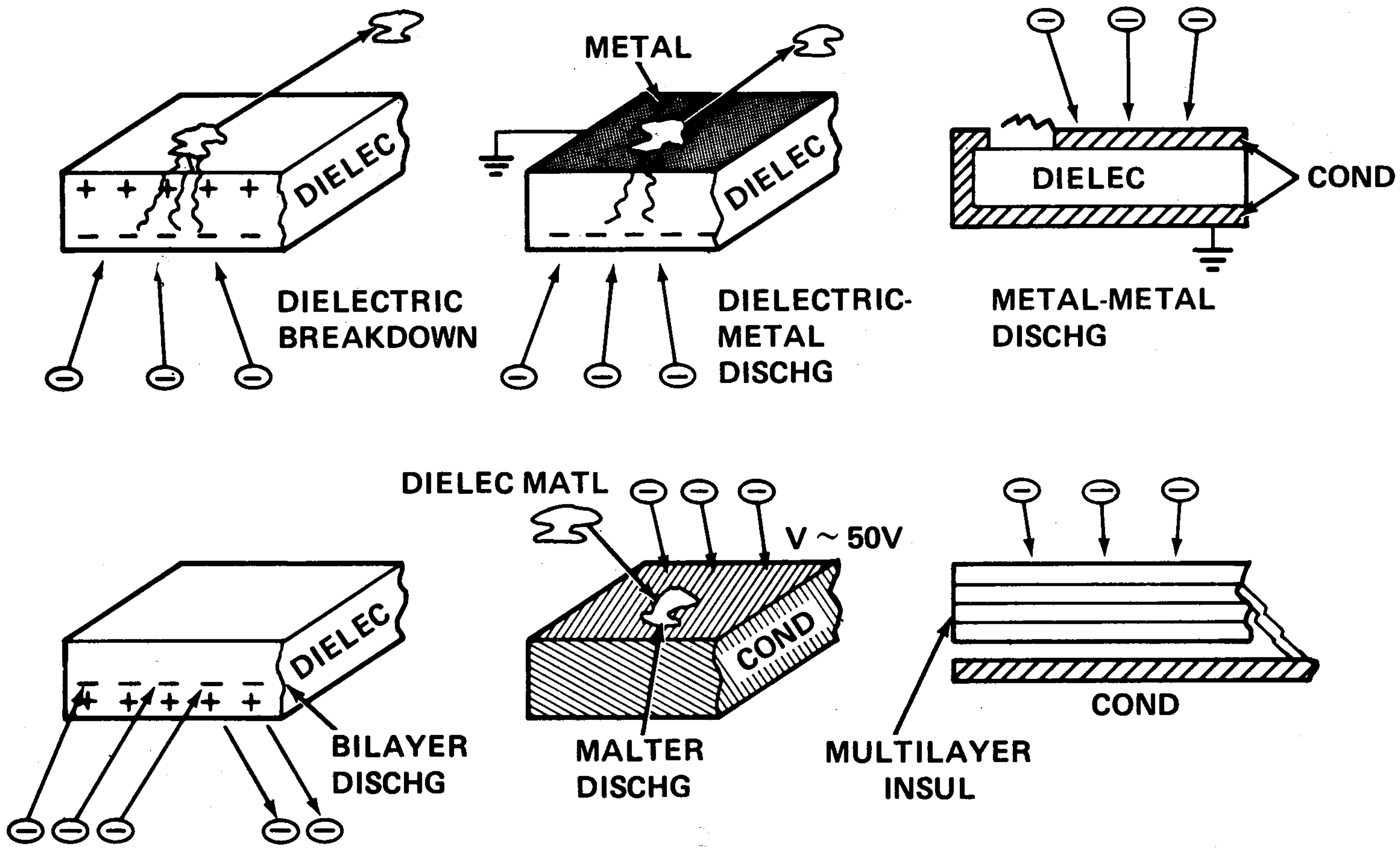


Figure 5

DIFFERENTIAL CHARGING/PLASMA INTERACTIONS (ELECTRICALLY POWERED SYSTEMS) (Figure 6)

Magnetic and electric field effects in electrically-powered systems may produce higher charge differentials than would exist in similar passive systems.

Many current loops are incorporated in large area solar cell arrays and active antennas. The resulting magnetic fields could focus or deflect electrons at various points throughout upper and lower surfaces. Distributed high voltage sections could accelerate electrons from a discharge at one point, producing increasing numbers of secondaries, and lead to a discharge avalanche.

Conductors exposed to the space plasma in electrically powered systems will attract electrons or ions depending on the conductor voltage with respect to the plasma. This particle flow constitutes a current loss through the plasma. The magnitude of this current is a function of electrical configuration, voltage level, exposed conductor area, and plasma characteristics.

Electric fields will attract charged particles to the surface of insulated conductors, thereby increasing the voltage gradient across the insulator. If insulator breakdown occurs, electrons may stream through the pinhole created by the rupture resulting in further material erosion and increased current loss.

For systems containing waveguide antennas, multipactor discharge is an additional concern; this phenomenon can occur if electrons enter the waveguide. Electrons within the waveguide will experience acceleration at the transmission frequency which will generate secondary electrons and create an electron plasma within the waveguide.

DIFFERENTIAL CHARGING/PLASMA INTERACTIONS (ELEC POWERED SYSTEMS)

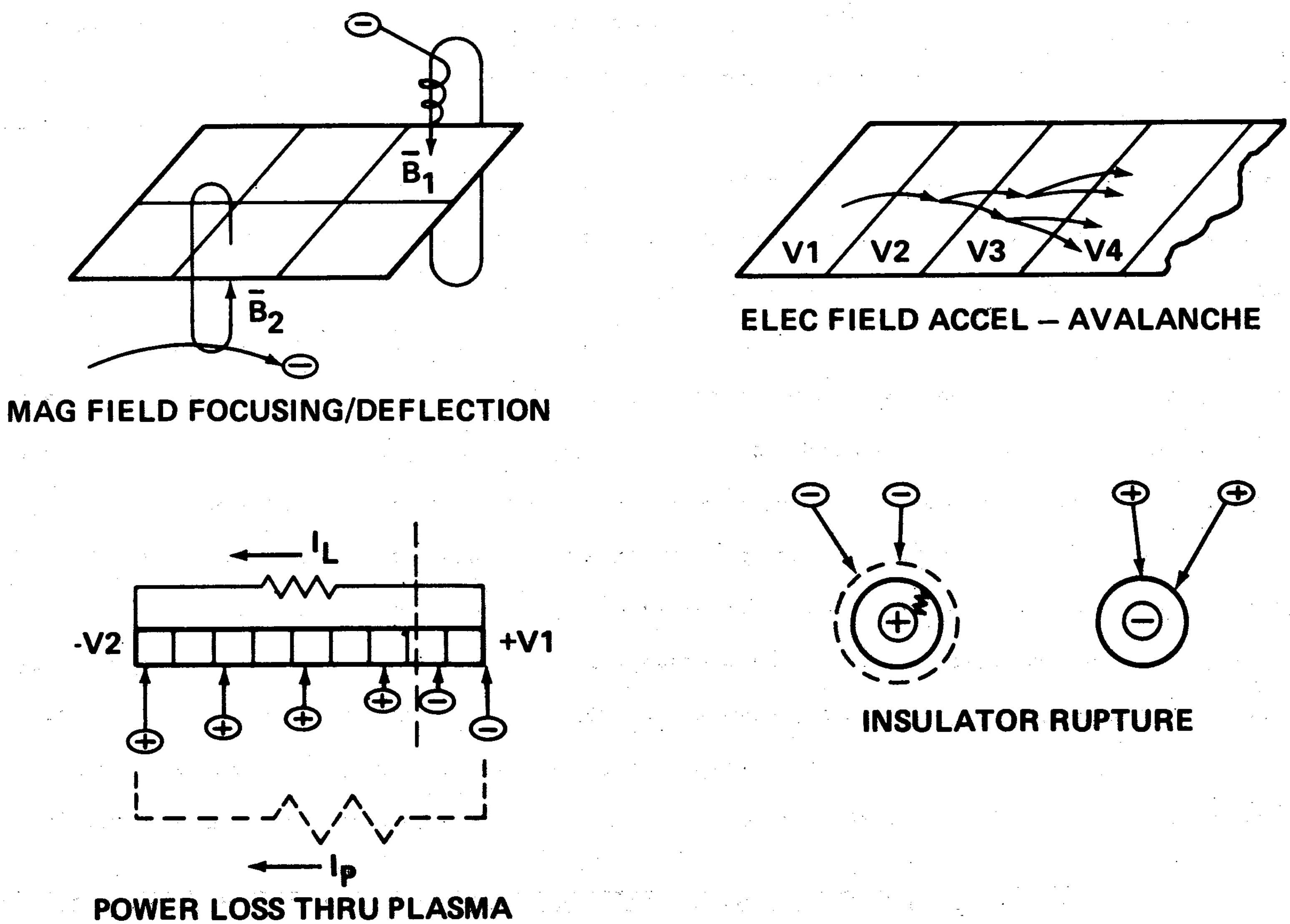


Figure 6

ELECTRICAL CHARGE/DISCHARGE EFFECTS (Figure 7)

Repeated electrical discharges can degrade optical properties of dielectric and metal surfaces used for thermal control, reflectors, solar cell covers, lenses/sensors, etc. These surfaces will degrade further from deposition of discharge contaminants.

Material breakdown could produce corrosive agents which accelerate erosion of integral metal conductors and electrical connections in the discharge vicinity.

Insulation pinholes can result from material breakdown and lead to shorting of electrical elements. All discharges will produce electromagnetic pulses resulting in possibly severe interference or electrical system voltage transients.

ELECTROSTATIC CHARGE/DISCHARGE EFFECTS

- DIELECTRIC CRAZING, DARKENING, EMBRITTLEMENT
- METAL CRAZING/VAPORIZATION
- COMPOSITE STRUCTURE BREAKDOWN/DAMAGE
- CONTAMINANT DEPOSITION
- CORROSIVE MATERIAL PRODUCTION
- EROSION/CORROSION OF THIN METAL CONDUCTORS
- INSULATION BURNTHRU/SHORTING
- EMI/VOLTAGE TRANSIENTS

Figure 7

PLASMA INTERACTIONS (Figure 8)

Plasma coupling currents and power losses could be significant for high voltage systems in LEO. Magnetic and electric fields associated with high power/voltage systems will influence local charge levels and discharge effects. These fields might be shaped where possible to reduce charging.

Coupling currents, system-generated fields, geomagnetic fields and accelerated ions will induce forces and torques which could impact structural, attitude/control/stationkeeping design requirements. Ions accelerated by high voltage systems will also increase surface sputtering and radiation damage.

If ion thrusters are used for transportation/control, thruster particles could be attracted to charged spacecraft surfaces, contributing to local spacecraft charging and contamination and reducing thruster effectiveness.

Electron multipacting can result in transmission power loss, shorting or damage to waveguide antennas.

PLASMA INTERACTIONS

- POWER LOSS THRU PLASMA
- MAGNETIC/ELECTRIC FIELD FOCUSING/ACCELERATION-AVALANCHE
- INDUCED FORCES/TORQUES
- INCREASED ION SPUTTERING/RADIATION DAMAGE
- ION THRUSTER CHARGING CONTRIBUTION, REDUCED THRUSTER EFF

Figure 8

LSS CHARACTERISTICS OF MAJOR CONCERN (Figure 9)

Many advanced missions (e.g., communications and surveillance) require very large area dielectric surfaces or coatings for antennas, optics, control skins, etc. These spacecraft may also include large solar cell arrays to satisfy high system/payload power requirements. Electrical discharges between differentially charged dielectrics, metals, and composites may seriously affect system performance and life. Additionally, very large, lightweight structures may be affected by disturbance forces and torques resulting from dynamic plasma interactions.

The synergistic effects of prolonged exposure to the space environment and repeated electrical discharges on material properties are of major importance and are among the greatest unknowns.

Multikilowatt power systems incorporate high current loops, and possibly high voltages, which could compound differential charging and plasma effects.

Full size or even large area ground testing under simulated combined environments may be prohibitive, thereby requiring accurate scaling techniques and large scale flight experiments. There is currently no activity which specifically addresses LSS requirements.

LSS CHARACTERISTICS OF MAJOR CONCERN

- MANY COMPOSED OF VERY LARGE AREA DIELECTRICS WITH INTEGRAL THIN CONDUCTORS – SUSCEPTIBLE TO DAMAGE
- EXTENSIVE USE OF COMPOSITE MATERIALS
- LARGE, LOW DENSITY STRUCTURES – INDUCED FORCES/TORQUES
- LONG LIFE REQUIREMENTS – MATERIAL AGING EFFECTS
- MAY INCLUDE HIGH POWER/VOLTAGE NETWORKS
  - CURRENT COUPLING/STABILITY
  - MAGNETIC/ELECTRIC FIELD FOCUSING/ACCEL
  - ECLIPSE & LOAD TRANSIENT EFFECTS
- LARGE SCALE EFFECTS UNKNOWN
  - CHARGE PROFILES/DISCHARGE MECHANISMS
  - EFFECTIVENESS OF CHARGE CONTROLS
  - PLASMA SHEATH FORMATION/CHARACTERISTICS
  - $\bar{B} \times \bar{v}$ & WAKE EFFECTS

Figure 9

DEPLOYABLE ANTENNA (Figure 10)

An example of a large spaceborne antenna is shown. This deployable design could scale to a 300m diameter with over 70,000m^2 projected surface area.

The antenna consists of several gores, the upper and lower planes of which are entirely composed of dielectrics (kapton) and thin metallic (aluminum, copper) elements. The ground plane is a metal mesh. Separation between ground and antenna planes is 3-25 cm. Kapton - copper delay lines interconnect antennas and ground planes. The gores are attached to a rim which is supported from the hub with upper and lower stays. The rim and stays are of graphite-epoxy composite; the remainder of the structure is aluminum alloy.

In an active array configuration, low power r.f. amplifiers, switches, and digital chips are distributed throughout the lower antenna plane. The feed system (not shown) is located on a mast deployed from the center cannister. Up to 50 kilowatts of electrical power could be supplied by a large solar cell array mounted on the same mast. Multiple current loops, connected to the common ground plane, feed the active elements in the lower antenna plane.

Electrostatic charging and discharges along, through, and between the dielectric-metallic gore layers and delay lines could cause material damage and antenna malfunction.

Another version of this deployable structure incorporates aluminized kapton gores which form a large sunlight reflector. The solar cell array and subsystem package is located at the hub in this case. Degraded surface reflectivity and structural properties of the substrate resulting from electrical discharges are of primary concern with systems of this type.

Possible physical and optical degradation of solar cell arrays and thermal control surfaces is also a major consideration for any large, electrically powered system.

DEPLOYABLE ANTENNA

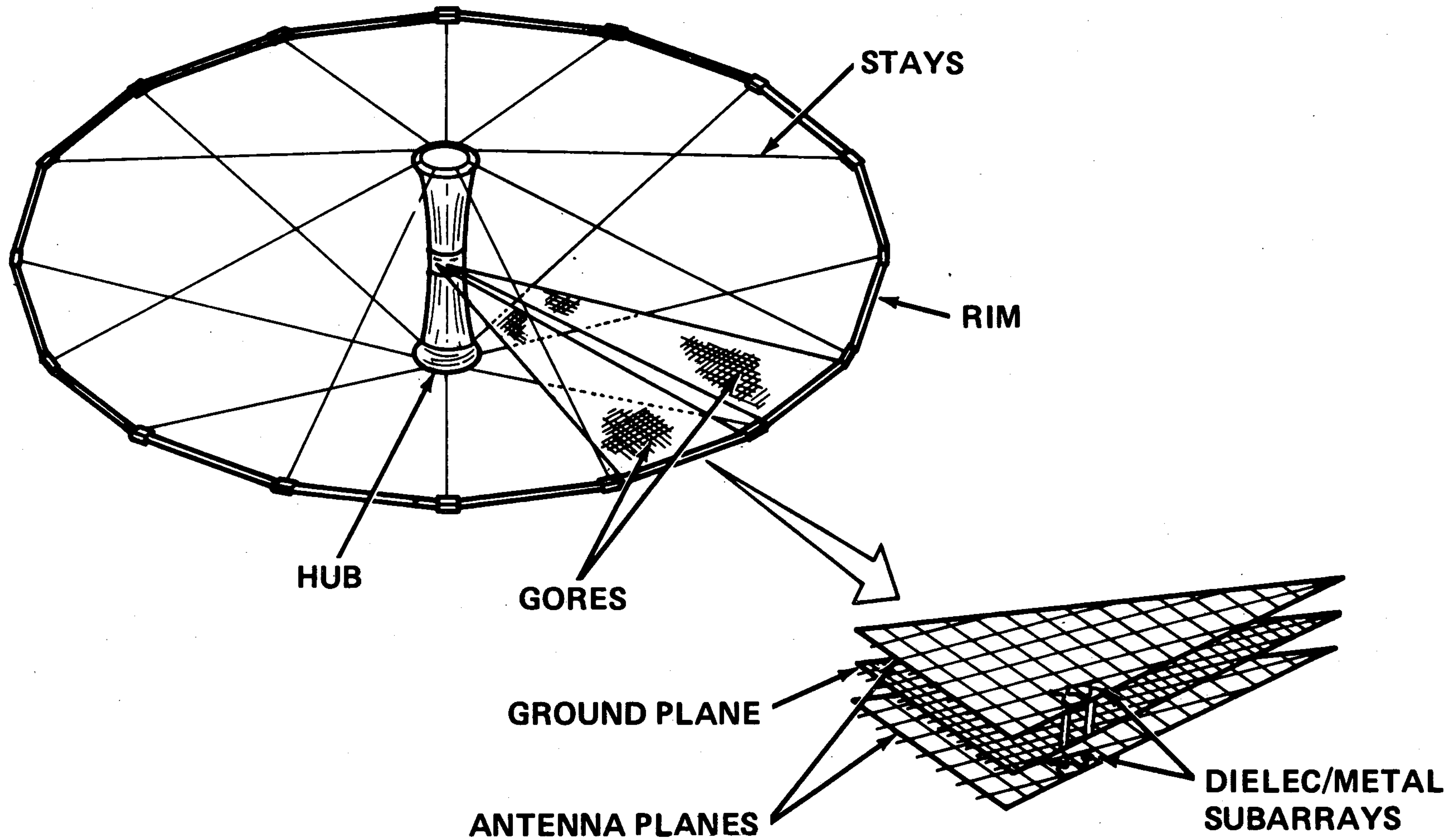


Figure 10

<u>COMMUNICATIONS PLATFORM</u> (Figure 11)

A concept for an advanced communications platform is shown. This design is constructed on orbit and consists of three multifunction systems, coupled in a platform, measuring 140M x 61M x 68M not including the solar cell arrays. The total projected surface area of the three lens antennas is 6100 M^2.

The lens aperture structure consists of hexagonal panels which are joined at the corners with a connection plate. The panels are of plastic honeycomb construction with photoetched, crossed-dipole conductors on the outer surfaces (antenna planes). Conductive sheets, bonded to the opposite honeycomb faces, form the ground plane. Antenna and ground planes are interconnected with metallized dielectric delay lines.

Support beams are graphite-epoxy with other structural elements of aluminum alloy. The feed/subsystem modules are of aluminum with some thermal control surfaces.

Power requirements for the communications and other subsystems could range from 80 to 100 kilowatts, requiring very large area solar cell arrays. Advanced solar cell arrays might incorporate thin, continuous, dielectric substrates and covers as shown in the detail. Dielectric breakdown and arcing in the vicinity of thin solar cell grids and interconnects could cause damage and output power degradation. The concerns regarding electrostatic charge/discharge effects are similar to those for the deployable antenna; long-term integrity and performance of dielectric and conductive elements and optical surfaces is the major concern.

COMMUNICATIONS PLATFORM

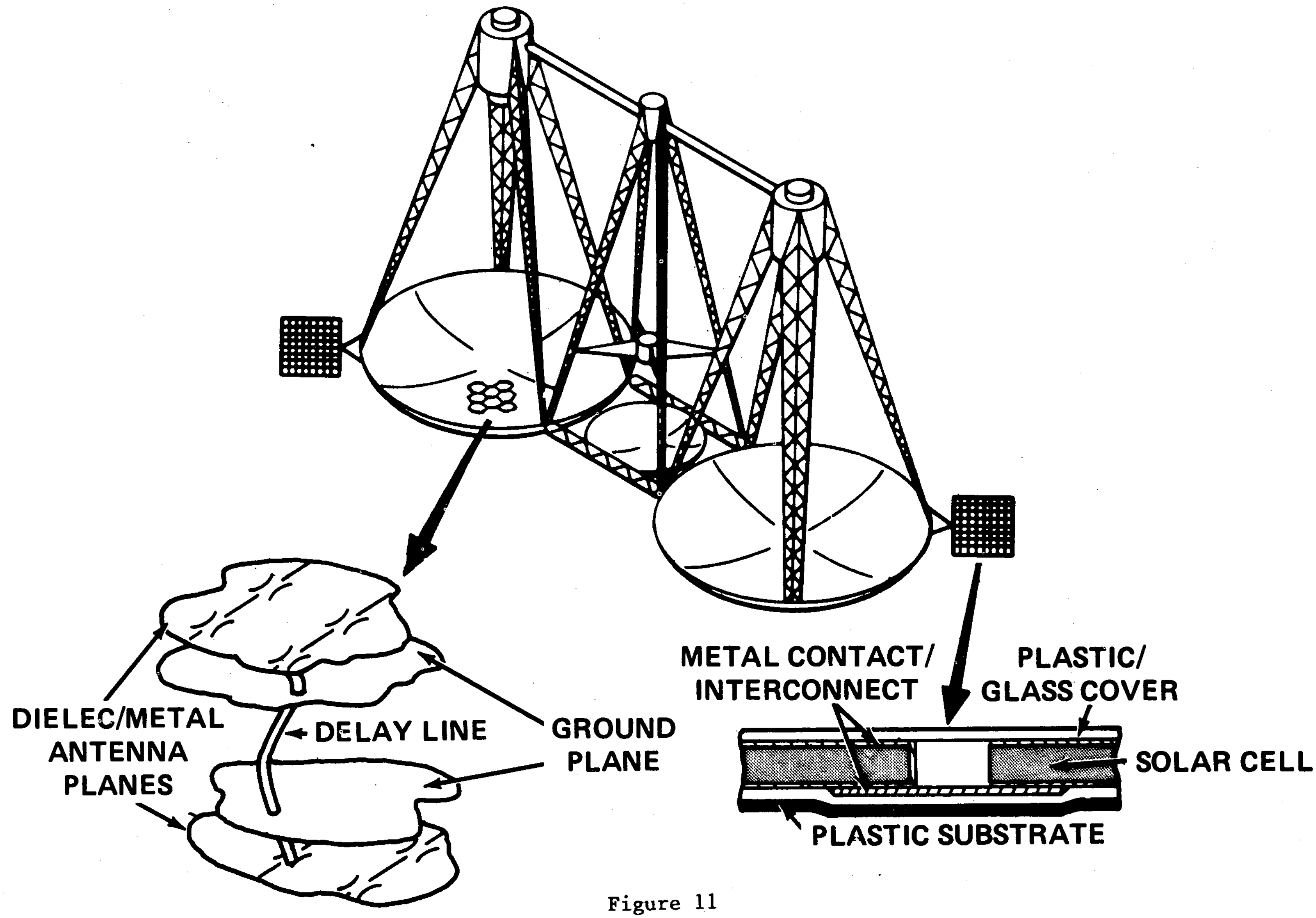


Figure 11

SPACECRAFT CHARGING INVESTIGATION PROGRAM (Figure 12)

A cooperative NASA/Air Force five-year program was initiated in 1976 to investigate spacecraft charging. The program plan includes environment definition, materials evaluation/development, test facility improvement, development of analytical tools, and generation of spacecraft design requirements/test specifications.

GEO environment models are derived through orbital data from ATS and other previous satellites and from ISEE (International Sun-Earth Explorer) satellites, the third of which will be launched by mid-1978. The SCATHA (Spacecraft Charging At High Altitudes) satellite will be launched in early 1979 and is dedicated to space plasma/charging evaluations.

A NASCAP computer program has been generated to predict charge levels on any conventional spacecraft but requires much improved plasma/materials response data. Charge response tests have been performed on some common space materials, but much has been exploratory in nature. Specific charge/discharge mechanisms have yet to be identified, and no work has been done on long-term effects. Materials development has concentrated on conductive coatings, paints, and alternate thermal blanket materials.

Several new activities must be initiated to evaluate the special problems posed by many Large Space Systems. These include development of plasma/charge/discharge models and NASCAP modifications and/or new programs for large, high power systems and accelerated materials ground or flight tests.

SPACECRAFT CHARGING/PLASMA INTERACTIONS— POSSIBLE CONTROL TECHNIQUES

- MAT'L SELECTION
- CONDUCTIVE COATINGS
- CONDUCTOR GRIDS/FRAMES
- "CONTROLLED" PIN HOLES
- DIELECTRIC SHAPING
- ELECTRON EMITTERS
- ELEC/MAG/RF FIELDS
- SHIELDS/GROUNDING
- LOW VOLTAGE SECTIONS
- CONDUCTOR INSULATION

Figure 12

SPACECRAFT CHARGING/PLASMA INTERACTIONS - POSSIBLE CONTROL TECHNIQUES (Figure 13)

Although all of the discharge effects and plasma interactions described previously are possible, their severity depends on the specific application. In addition the data presently available will not permit evaluations with confidence for most applications.

When estimates are made of the location and magnitude of charging/plasma effects on a particular design, one or more controls might be used to prevent discharge/interaction problems.

Relatively simple (passive) controls include selection of suitable materials, designed charge leakage paths, and elimination of high electrical stress locations.

Use of active controls (electron emitters, shaped electromagnetic fields) has been shown to reduce surface charging, but large scale designs and performance estimates have not yet been developed.

Good electrical design practices must be followed to minimize breakdown/arcing problems in high voltage systems; special design requirements will probably have to be developed to cope with compounding effects of space charging. Also, maximum system voltages may have to be limited to control plasma acceleration/power loss problems.

The LDEF (Long Duration Exposure Facility) will include some charge response tests in LEO; others may be planned for recovery from GEO or elliptical orbits. The SPHINX (Space Plasma, High Voltage, Interaction Experiment) was fully ground tested but failed to orbit. These experiments, if modified and launched on early Shuttle flights, could provide valuable new data for LSS modelling. At least one large area GEO experiment will be required to measure LSS plasma sheath characteristics and verify scaling techniques. Once sufficient ground/flight materials effects data is generated and analytical predictions are verified, LSS design specifications can be developed.

SPACECRAFT CHARGING INVESTIGATION PROGRAM

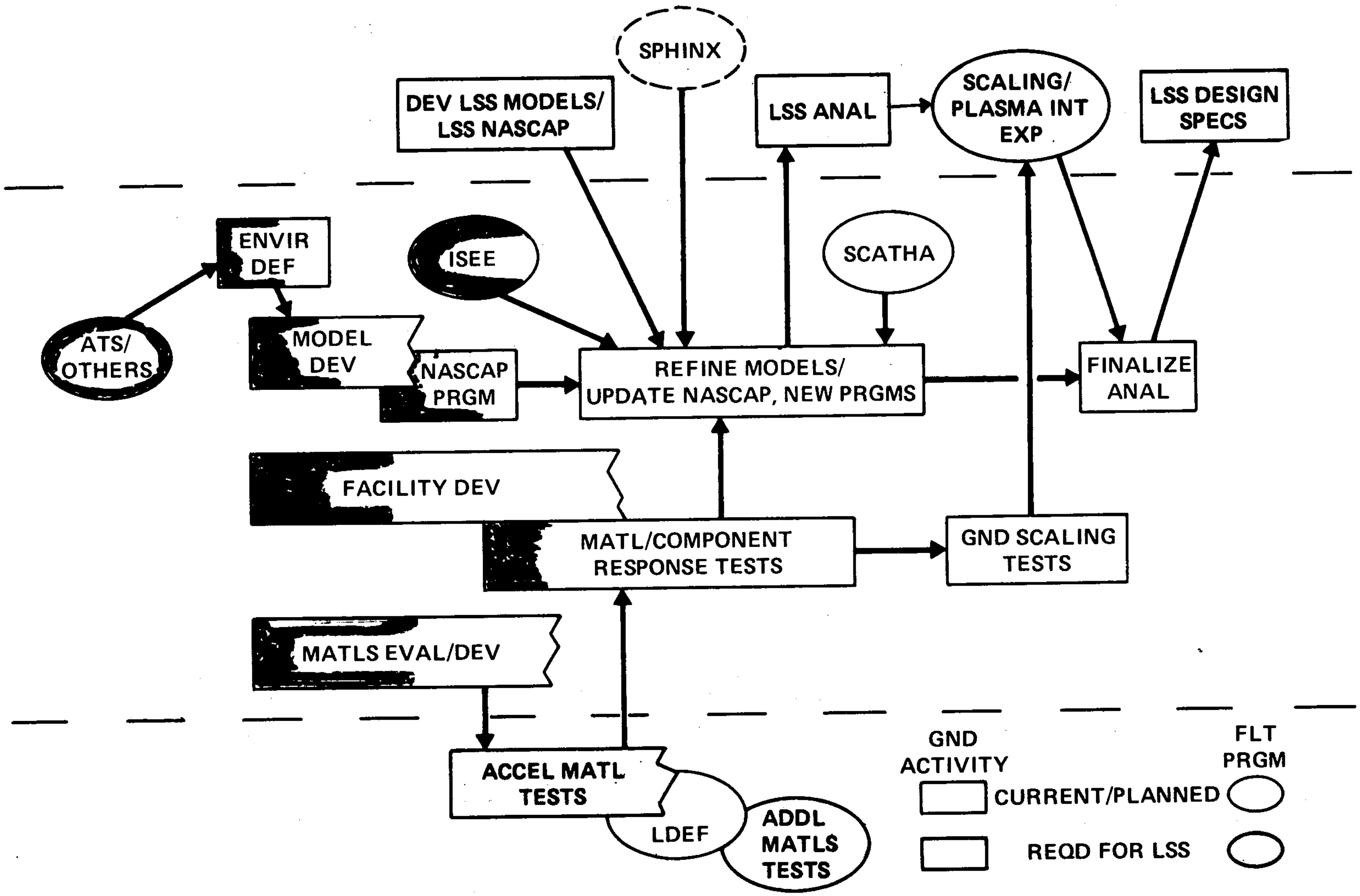


Figure 13

SCALING/PLASMA INTERACTION FLIGHT EXPERIMENT CONCEPTS (Figure 14)

A large area, scaling/plasma interaction flight experiment is an obvious requirement in an LSS-oriented spacecraft charging investigation program. Although it is premature to define specific requirements, two distinct concepts should be considered.

The first is a low power (less than one kilowatt) satellite which deploys (rollout/foldout) a large area blanket composed of material/component test samples and sensors. Exposed test area and orientation would be variable to provide parametric scaling, illumination, and geomagnetic data. Experiments might include plasma sheath formation/stability/uniformity, charge response, passive/low - power active controls, and high voltage interactions.

A power module could be used to support a high power (10-20 kilowatt) experiment which incorporates a variable power network and a simple, instrumented, deployable structure. In addition to the tests listed above, an experiment of this type could evaluate magnetic field, current coupling, eclipse, and load transient effects on plasma sheath characteristics, charge response, and structural dynamics. The higher-powered electromagnetic charge controls could also be evaluated.

The maximum dimension of these test articles might be anywhere from 100 m to 1 km. Mission times would be set to include a reasonable number and variety of substorm events, or the experiments could be flown for several years to derive additional aging effects data.

SCALING/PLASMA INTERACTION FLIGHT EXPERIMENT CONCEPTS

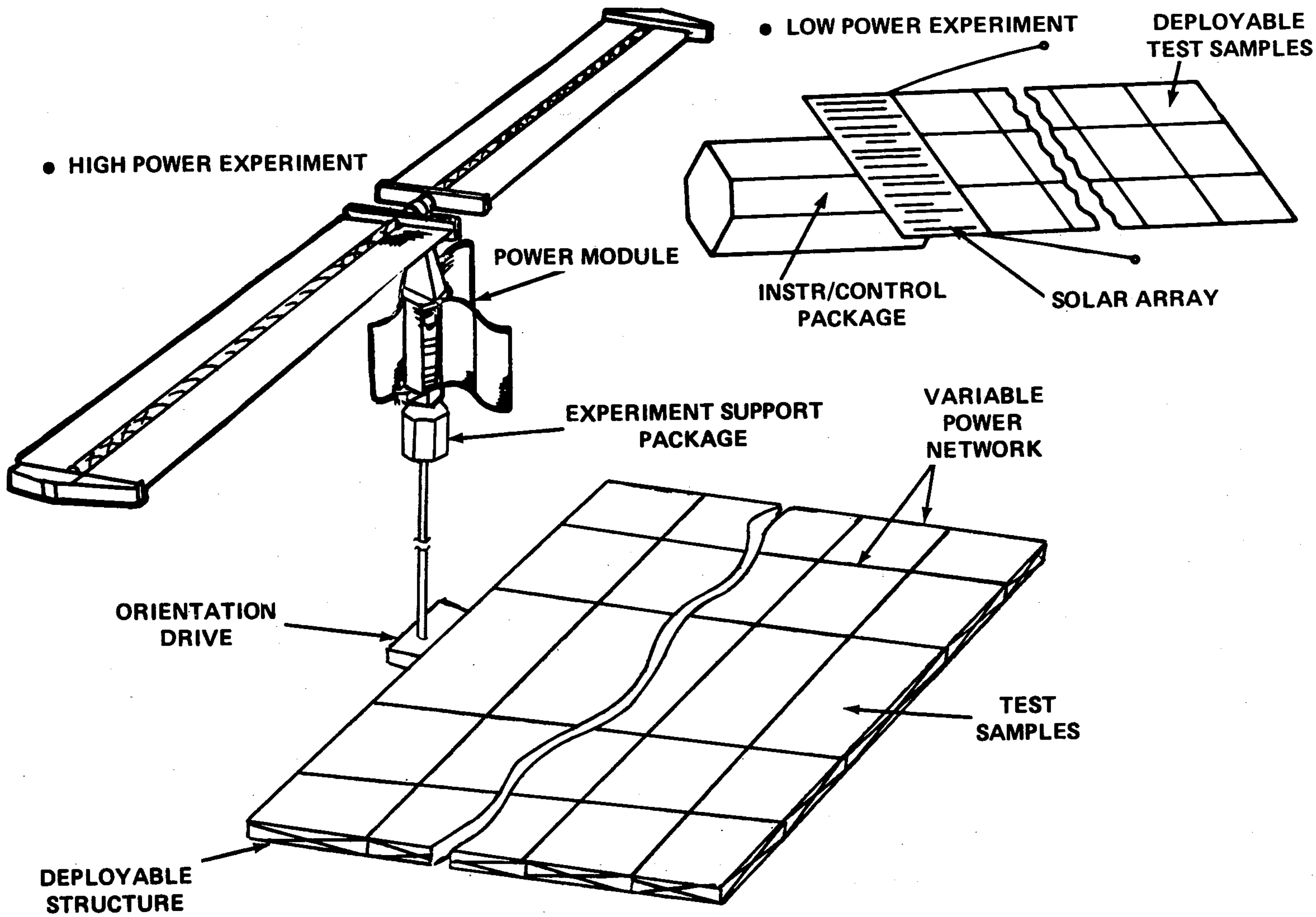


Figure 14

CONCLUSIONS (Figure 15)

Potentially serious consequences may arise from electrostatic charging and plasma interactions with Large Space Systems. Little is known about the specific mechanisms which must be controlled, and almost no data exists on large-scale or long-term environmental effects.

Although current programs promise answers to many fundamental questions and near-term solutions for conventional satellites, they do not address specific LSS problems.

Since few estimates have been made of the magnitude of LSS charging problems, the list of potentially damaging effects is impressive. Specific LSS charging studies could, however, show that many problems can be eliminated with relatively simple controls with minor impact on spacecraft designs.

Now is the time to address these LSS charging/plasma interaction problems; their impact must be assessed before LSS programs proceed much beyond the conceptual design phase.

CONCLUSIONS

- ELECTROSTATIC CHARGING/PLASMA INTERACTIONS MAY HAVE SERIOUS IMPACT ON PERFORMANCE/COST OF LSS IN GEO
- CURRENT STUDIES DO NOT ADDRESS SPECIFIC LSS PROBLEMS
- LONG TERM ENVIRONMENT/ELECTRICAL DISCHARGE EFFECTS ON MATERIALS MOST CRITICAL DATA REQUIREMENT
- MANY PROBLEMS MIGHT BE ELIMINATED WITH RELATIVELY SIMPLE CHARGE CONTROLS

Figure 15

RECOMMENDATION (Figure 16)

A detailed LSS charging investigation program plan should be developed now; major LSS-oriented activities should be integrated into the current NASA/Air Force program.

The first steps in the program will include preliminary modelling/analysis of typical large systems which appear most susceptible to charging problems. Development of accelerated ground and/or flight tests to determine materials aging effects for long duration LSS missions is a most important early activity. Maximum use should be made of LDEF and SPHINX experiments to derive early material response/plasma interaction data.

Emphasis should be placed on development of flight experiments to investigate large scale plasma characteristics/interactions and verify analytical models.

RECOMMENDATIONS

- PLAN & IMPLEMENT A LSS CHARGING INVESTIGATION PROGRAM
  - INTEGRATE LSS IN CURRENT NASA/AF STUDIES
  - DEVELOP CHARGING/INTERACTION MODELS FOR LSS OPTIONS; MODIFY NASCAP PROGRAM & ESTIMATE IMPACT
  - ANALYZE ACCELERATED MATERIALS TEST METHODS; DESIGN GROUND/FLIGHT TESTS
  - DEVELOP LDEF CHARGING EXPERIMENTS; CONSIDER ELLIPTICAL OR GEO LDEF FLIGHTS
  - UPDATE SPHINX EXPERIMENTS; FLY IN LEO/ELLIPTICAL ORBIT
  - DEVELOP SCALING/PLASMA INTERACTION FLIGHT EXPERIMENTS

Figure 16

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER	NAME:
		DATE:

NASA/DOD/INDUSTRY SEMINAR ON LARGE SPACE SYSTEM TECHNOLOGY

"MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS"

BY

RAYMOND L. GAUSE, CHIEF, ENGINEERING PHYSICS DIVISION,
MATERIALS AND PROCESSES LABORATORY, MARSHALL SPACE FLIGHT CENTER

ABSTRACT:

Long Life Large Space Systems (LSS) are envisioned as NASA primary space missions in the 1985 to 2000 year time frame. To make such systems a reality will require the advancement of materials technology on several fronts. For example, to achieve the goal of a 20 to 30 year operational life in a geosynchronous earth orbit environment will require the use of materials which are highly resistant to degradation of critical functional properties due to electron and proton particulate radiation and solar electromagnetic radiation. Of particular LSS importance is long term dimensional stability in a thermal/vacuum/radiation environment. Since the space environment, in general, can be considered a hostile one for many materials, the long term effects of this environment on materials and systems integrity and performance is of vital importance and must be evaluated in order to design a space system that will meet these functional goals. Thus, an LSS materials data base must be available to the designer to permit the selection of materials which are "space stable". The purpose of this paper is to discuss LSS materials requirements in terms of types of materials, critical properties, and environmental stability and outline the materials technology development that will be needed to meet these requirements.

(Figure 1)

THE TOPICS TO BE COVERED BY THIS PRESENTATION ARE DEPICTED ON THIS CHART. IN ORDER TO ESTABLISH THE MATERIALS TECHNOLOGY DEVELOPMENT REQUIRED FOR LSS, IT IS NECESSARY TO FIRST ESTABLISH GENERAL MATERIALS REQUIREMENTS THAT MUST BE MET TO SATISFY PROGRAM GOALS. THEN, IN ORDER TO SCOPE THE TECHNOLOGY EFFORT, THE RESULTS OF STUDY EFFORTS MUST BE ASSESSED TO DEFINE THE TYPES OF MATERIALS THAT WILL BE NEEDED, HOW THEY WILL BE USED, AND THE PROPERTIES THAT WILL BE CRITICAL FOR THE ANTICIPATED APPLICATIONS. SINCE THESE MATERIALS WILL BE EXPOSED TO THE SPACE ENVIRONMENT FOR A PERIOD OF TIME LONGER THAN ANY OTHER MATERIALS IN THE HISTORY OF THE SPACE PROGRAM, IT IS EXTREMELY IMPORTANT TO EVALUATE THE EFFECTS OF THIS SPACE EXPOSURE ON MATERIALS PERFORMANCE. WITH THIS BACKGROUND IT IS THEN POSSIBLE TO DEFINE A TECHNOLOGY PROGRAM TAILORED TO DEVELOP THE HIGH PERFORMANCE MATERIALS THAT WILL BE NEEDED FOR LSS.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS	NAME: DATE:

TOPICS

- MATERIALS REQUIREMENTS FOR LARGE SPACE SYSTEM (LSS)
- TYPICAL LSS MATERIALS, APPLICATION, AND CONTROLLING PROPERTIES
- ENVIRONMENTAL CONSIDERATIONS
  - LEO
  - GEO
- MATERIALS TECHNOLOGY DEVELOPMENT FOR LSS

Figure 1

(Figure 2)

IT IS OBVIOUS THAT ONE OF THE PRIME REQUIREMENTS FOR LSS MATERIALS IS THAT THEY BE LIGHTWEIGHT WHICH ALSO IMPLIES A HIGH STRENGTH TO WEIGHT RATIO. LOW COST IS IMPORTANT WHEN CONSIDERING THE SHEER SIZE OF THE STRUCTURES AND THE AMOUNT OF MATERIAL THAT WILL BE REQUIRED. IN ADDITION, THE MATERIALS SHOULD BE EASILY PROCESSED. THIS MEANS NOT ONLY MINIMIZING LABOR AND TIME BUT ALSO THE TOOLING AND POWER REQUIRED FOR PROCESSING. THE LAST REQUIREMENT LISTED PERTAINS TO THE SPACE DURABILITY OF THE LSS MATERIALS AND IS THE MOST SIGNIFICANT AS FAR AS MATERIALS TECHNOLOGY IS CONCERNED. NO LONG TERM ENVIRONMENTAL DATA ARE AVAILABLE WITH WHICH TO ASSESS MATERIALS PERFORMANCE. SINCE MAINTENANCE AND REPAIR OF THESE STRUCTURES AND SYSTEMS WILL BE DIFFICULT AND EXPENSIVE, IT IS IMPERATIVE THAT THE BASIC MATERIALS USED MAINTAIN THEIR CRITICAL PROPERTIES WITHIN SPECIFIED LIMITS FOR THE BASELINED LIFETIME OF 20 TO 30 YEARS.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS	NAME: DATE:

MATERIALS REQUIREMENTS

- LIGHT WEIGHT
- LOW COST
- EASILY PROCESSED
- LONG TERM RETENTION OF PROPERTIES IN SPACE ENVIRONMENT (20 TO 30 YEARS)

Figure 2

(Figure 3)

BASED ON A REVIEW OF THE LSS FOCUS MISSIONS AND ASSOCIATED HARDWARE, GENERAL MATERIALS CATEGORIES, TYPICAL APPLICATIONS, AND CONTROLLING PROPERTIES FOR THESE APPLICATIONS HAVE BEEN DEFINED AS SHOWN IN THIS CHART. IT IS NOT INTENDED THAT THIS CHART BE ALL INCLUSIVE BUT THAT IT BE ILLUSTRATIVE OF THE BREADTH OF THE MATERIALS DATA BASE THAT WILL BE NEEDED TO SUPPORT THE LSS.

ORGANIZATION:

MARSHALL SPACE FLIGHT CENTER

MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS

NAME:

DATE:

		CONTROLLING PROPERTIES				
MATERIAL TYPE	TYPICAL LSS APPLICATION	MECH	OPTICAL	ELECT	CoTE	OUTGASSING
COMPOSITES	STRUCTURAL MEMBERS	X			X	X
THIN GAUGE METALS	STRUCTURAL MEMBERS	X			X	
ADHESIVES	JOINTS	X				X
DIELECTRICS	ELECTRICAL/ELECTRONIC SYSTEMS	X		X		X
COATINGS	THERMAL CONTROL REFLECTORS	X	X			X
THIN FILMS	THERMAL BLANKETS MIRRORS	X	X		X	X
WIRE MESHES	ANTENNAE	X			X	
SEMICONDUCTORS	ELECTRONICS, SOLAR CELLS			X		
GLASSES	SOLAR CELL COVERS		X			

Figure 3

(Figure 4)

THE ELECTRON ENVIRONMENT FOR GEOSYNCHRONOUS ORBIT IS GIVEN BY VETTE'S MODEL AE4. THE INTEGRAL ELECTRON SPECTRA FOR L = 6.6 AT SOLAR MINIMUM IN EPOCH 1964 IS SHOWN WITH TIME AVERAGED VALUES USED TO AVERAGE OUT THE EFFECTS OF MAGNETIC STORMS.[1,2] THE ION ACCELERATOR FACILITY AT MSFC CAN PROVIDE ELECTRON IRRADIATIONS OVER THE ENERGY RANGE OF 0.35 MEV TO 2.5 MEV TO SIMULATE THIS ENVIRONMENT.

(1) SINGLEY, G.W. AND VETTE, J.I., THE AE4 MODEL OF THE OUTER RADIATION ZONE ELECTRON ENVIRONMENT. NSS DC 72-06, AUGUST 1972.

(2) THE EARTH'S TRAPPED RADIATION BELTS. NASA SP-8116, MARCH 1975.

INTEGRAL ELECTRON SPECTRA, GEOSYNCHRONOUS ORBIT

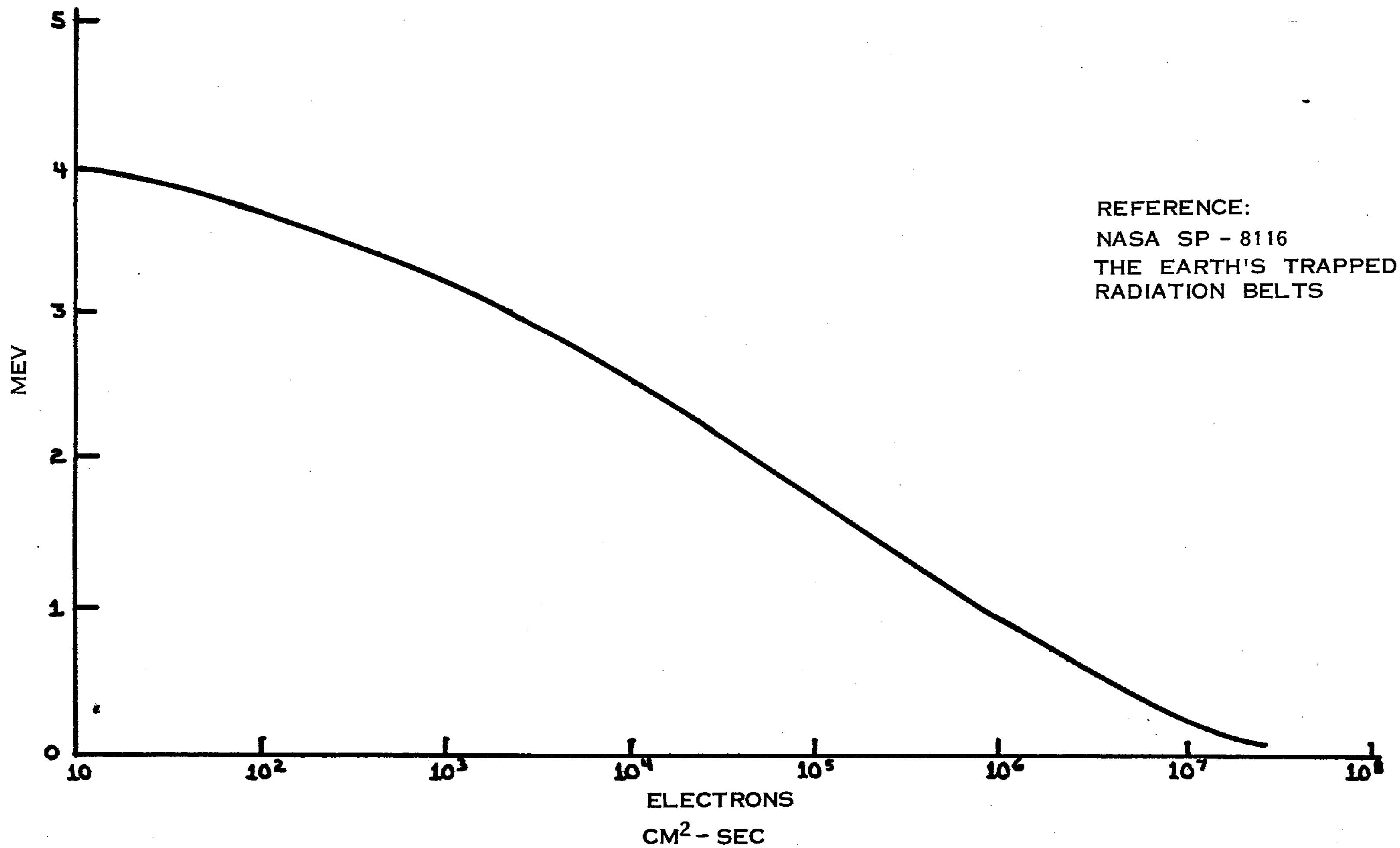


Figure 4

(Figure 5)

THE CUMULATIVE ELECTRON EXPOSURE IN GEOSYNCHRONOUS ORBIT IS SHOWN AS DERIVED FROM THE FIGURE FOR L = 6.6 FOR ELECTRONS WITH ENERGIES GREATER THAN 0.5 MEV. FOR SURFACE RADIATION DAMAGE EFFECTS, THE LOWER ENERGY ELECTRONS MUST BE INCLUDED AND THE TOTAL FLUENCE WOULD BE HIGHER THAN SHOWN. FOR A 20 TO 30 YEAR STAY TIME IN GEO, THE PREDICTED FLUENCE IS ON THE ORDER OF 3×10^{15} e/cm^2. THIS ELECTRON EXPOSURE CAN INDUCE SIGNIFICANT PROPERTY CHANGES IN MANY TYPES OF MATERIALS.

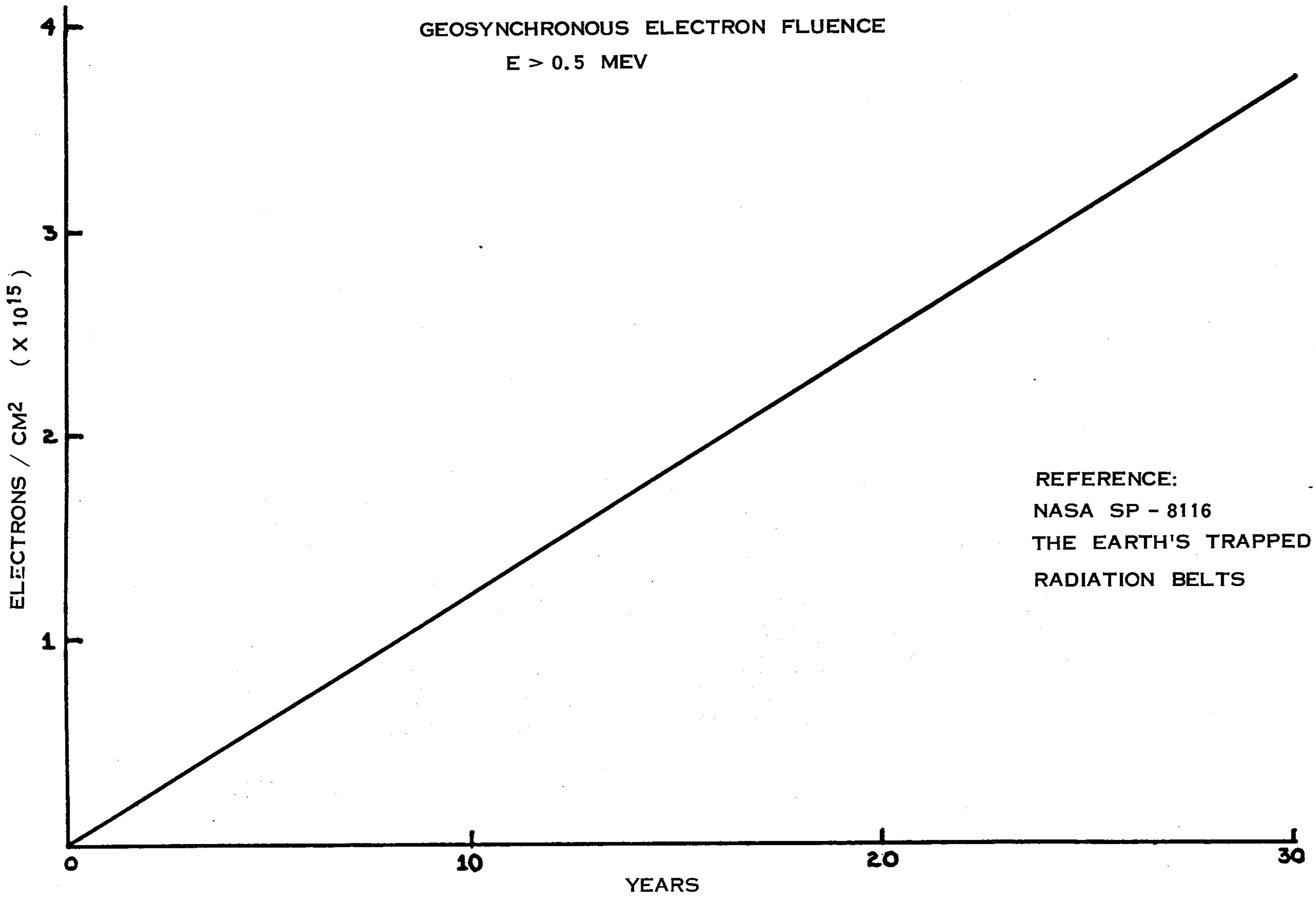

GEOSYNCHRONOUS ELECTRON FLUENCE
E > 0.5 MEV
ELECTRONS / CM2 (X 10^15)
4
3
2
1
0
10
20
30
YEARS
REFERENCE:
NASA SP - 8116
THE EARTH'S TRAPPED
RADIATION BELTS

Figure 5

(Figure 6)

THE TOTAL RADIATION DOSE RECEIVED BEHIND A 1 g/cm^2 THICK ALUMINUM SHIELD IS SHOWN.[1] FOR THIS THICKNESS, THE PRIMARY SOURCE OF RADIATION IS THE ELECTRON FLUENCE. FOR THICKNESSES OF MORE THAN 1.8 g/cm^2 OF ALUMINUM, THE PRIMARY RADIATION SOURCE WOULD BE BREMSSTRAHLUNG PHOTONS GENERATED BY THE ELECTRON FLUENCE'S INTERACTIONS WITH THE ALUMINUM ATOMS. SINCE THE BREMSSTRAHLUNG FLUX IS A FUNCTION OF THE ATOMIC NUMBER OF THE TARGET MATERIAL, THE RADIATION DOSE WITHIN A GEOSYNCHRONOUS SPACE STRUCTURE WOULD BE HIGHLY VARIABLE AND THEREFORE MUST BE CAREFULLY CALCULATED FOR EACH SPACE STRUCTURE AND/OR COMPONENT. FOR ILLUSTRATIVE PURPOSES, HOWEVER, USING THIS GRAPH, AN ELECTRONIC COMPONENT IN A 1 g/cm^2 ENCLOSURE WOULD RECEIVE A DOSE ON THE ORDER OF 2×10^5 RADS IN 20 YEARS. THIS DOSE IS SUFFICIENT TO SERIOUSLY DEGRADE THE PERFORMANCE OF SOME ELECTRONIC DEVICES.

(1) WRIGHT, J.J. AND FISHMAN, G.J., RADIATION ENVIRONMENT AND HAZARDS FOR A GEOSYNCHRONOUS SPACE STATION. NASA TMX-64983, FEBRUARY 1976

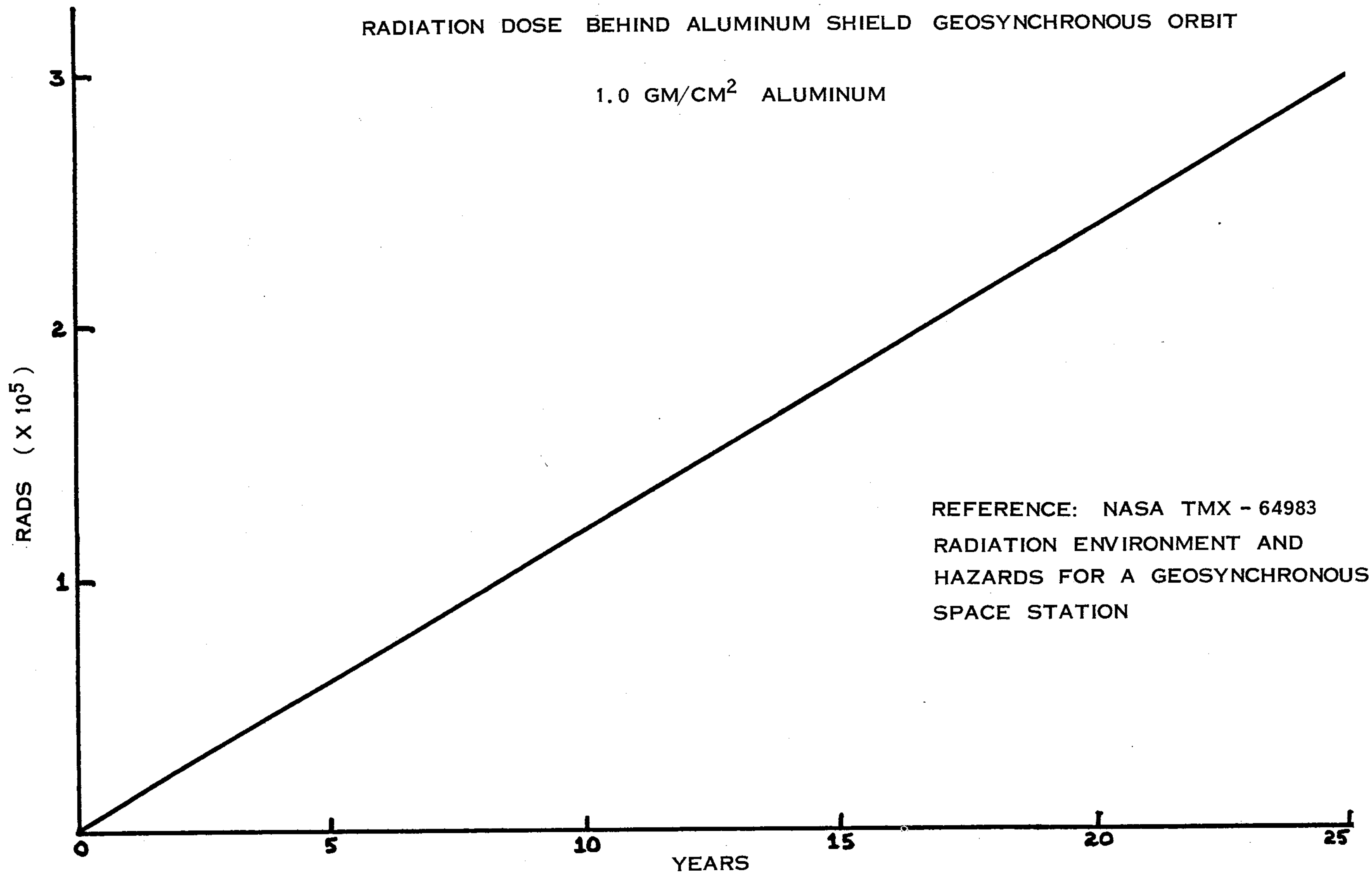

RADIATION DOSE BEHIND ALUMINUM SHIELD GEOSYNCHRONOUS ORBIT
1.0 GM/CM^2 ALUMINUM
RADS (X 10^5)
3
2
1
0
5
10
15
20
25
YEARS
REFERENCE: NASA TMX - 64983
RADIATION ENVIRONMENT AND
HAZARDS FOR A GEOSYNCHRONOUS
SPACE STATION

Figure 6

(Figure 7)

THE TOTAL ELECTRON FLUENCE AT 800 KM ORBIT AND 28.5° INCLINATION IS SHOWN.[1] THESE FLUENCES ARE ABOUT AN ORDER OF MAGNITUDE LESS THAN THOSE AT GEOSYNCHRONOUS ORBIT; HOWEVER, IT IS TO BE NOTED THAT THE PEAK FLUENCES LIE BETWEEN THESE TWO ORBITS AT ABOUT $L = 4$ TO $L = 5$.

(1) WATTS, J.W., JR., AND WRIGHT, J.J., CHARGED PARTICLE RADIATION ENVIRONMENT FOR THE SPACELAB AND OTHER MISSIONS IN LOW EARTH ORBIT, REVISION A. NOVEMBER 1976. NASA TMX-73358, NOVEMBER 1976.

ELECTRON FLUENCE AT 800 KM AND 28.5° INCLINATION

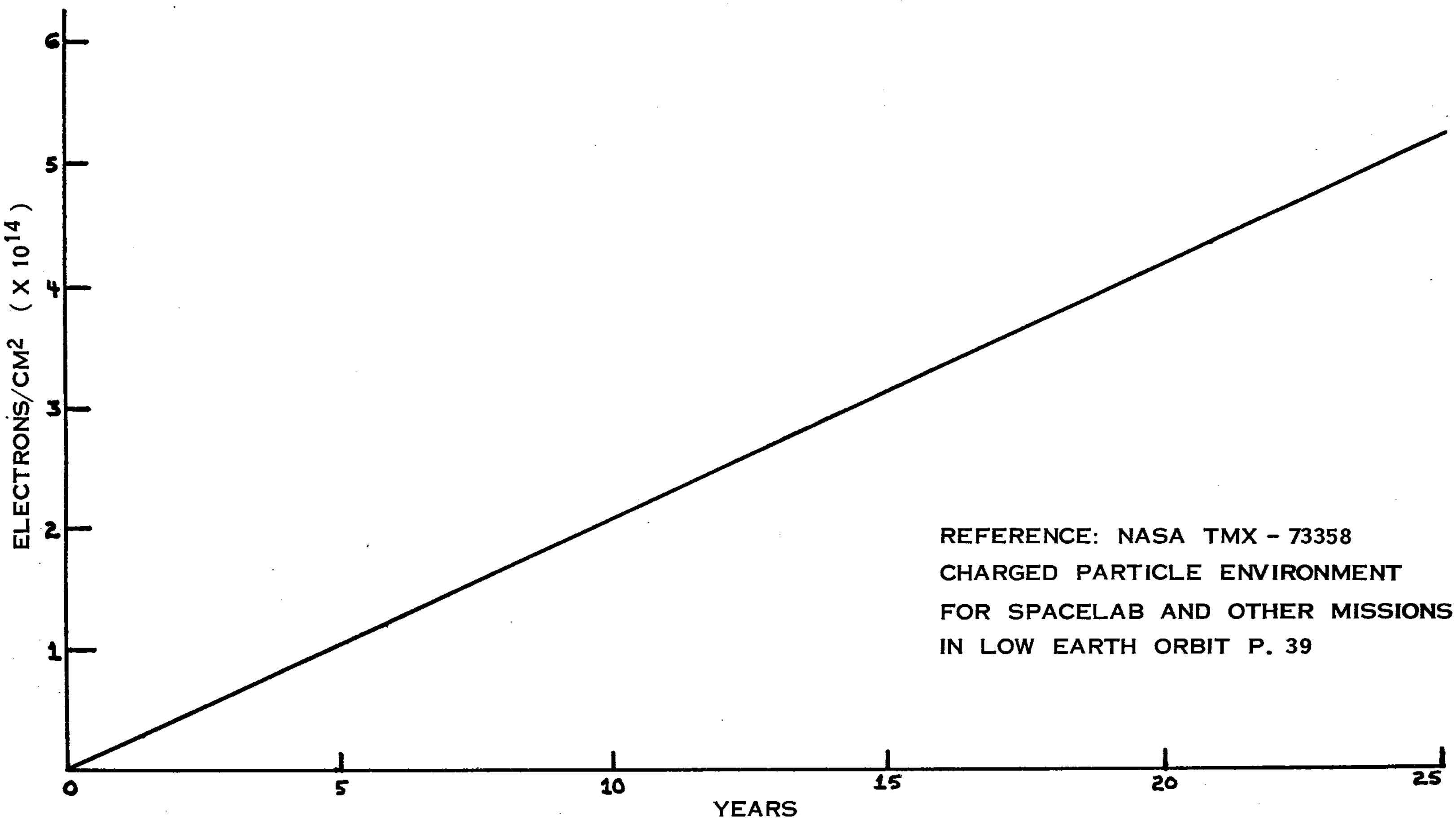


Figure 7

(Figure 8)

THE TOTAL ELECTRON FLUENCE AT A 200 KM ORBIT AND 28.5^{o} INCLINATION IS SHOWN. THIS FLUENCE IS ABOUT 4-1/2 ORDERS OF MAGNITUDE LESS THAN THAT AT THE PREVIOUS 800 KM ORBIT. AT THIS ALTITUDE,THE PROTON FLUENCE IN THE SOUTH ATLANTIC ANOMALY IS THE PRIMARY RADIATION SOURCE.

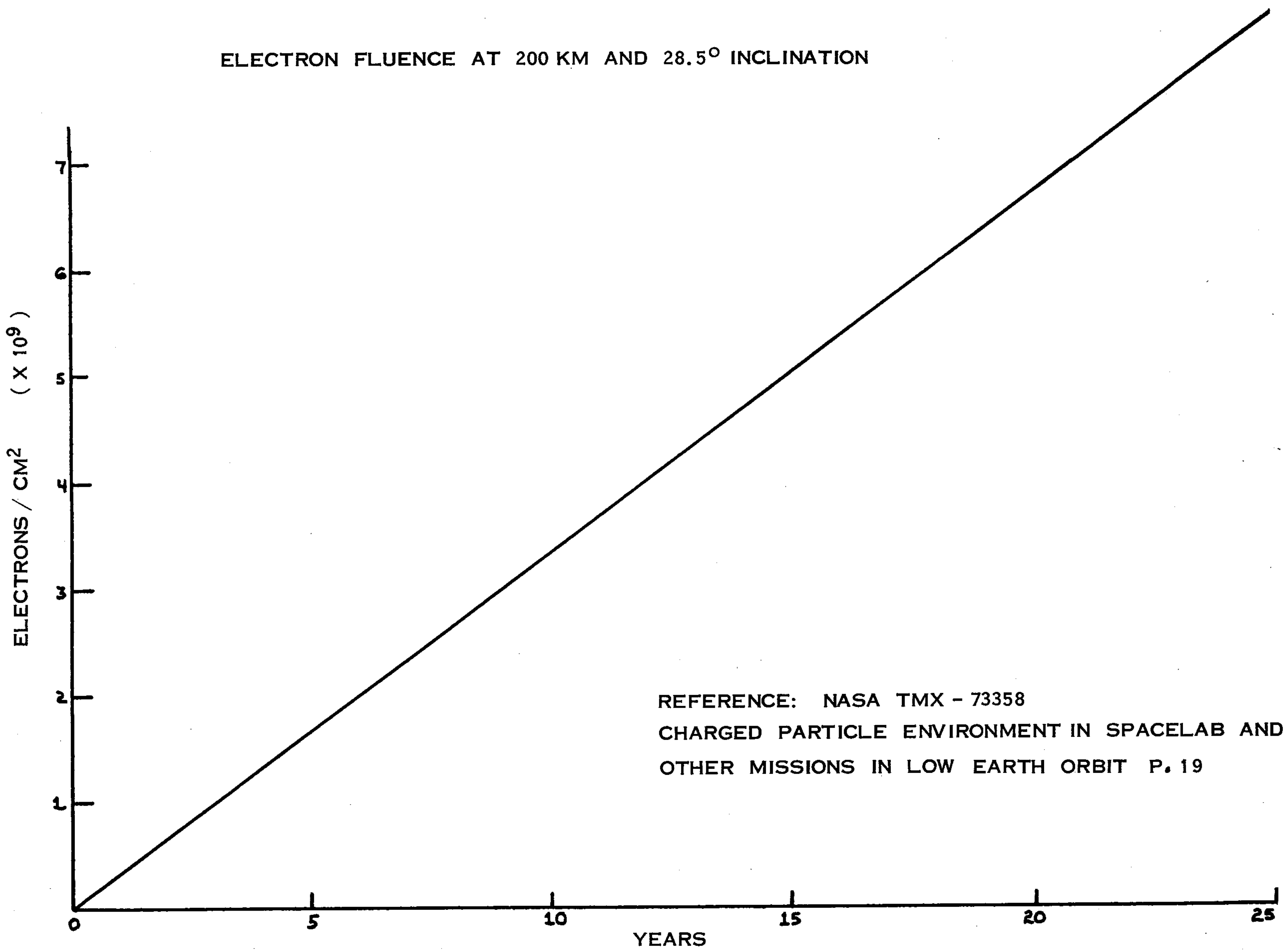

ELECTRON FLUENCE AT 200 KM AND 28.5° INCLINATION
ELECTRONS / CM2 (X 109)
7
6
5
4
3
2
1
0
5
10
15
20
25
YEARS
REFERENCE: NASA TMX - 73358
CHARGED PARTICLE ENVIRONMENT IN SPACELAB AND
OTHER MISSIONS IN LOW EARTH ORBIT P. 19

Figure 8

(Figure 9)

THE THICKNESS OF ALUMINUM AND NON-METALLIC COMPOSITE MATERIALS FOR LSS STRUCTURAL MEMBERS WILL RANGE FROM 0.010 TO 0.050 INCH. AS CAN BE SEEN FROM THIS CHART, ELECTRONS WITH AN ENERGY GREATER THAN APPROXIMATELY 0.3 MEV WILL HAVE SUFFICIENT RANGE IN THESE MATERIALS TO AFFECT THE BULK PROPERTIES, WHEREAS LOWER ENERGY ELECTRONS WILL PRIMARILY AFFECT SURFACE PROPERTIES. SINCE THE GEO ENVIRONMENT HAS A SIGNIFICANT FLUX OF ELECTRONS IN THIS ENERGY RANGE, BOTH THE SURFACE AND BULK PROPERTIES OF MATERIALS CAN BE EXPECTED TO BE AFFECTED IN GEO.

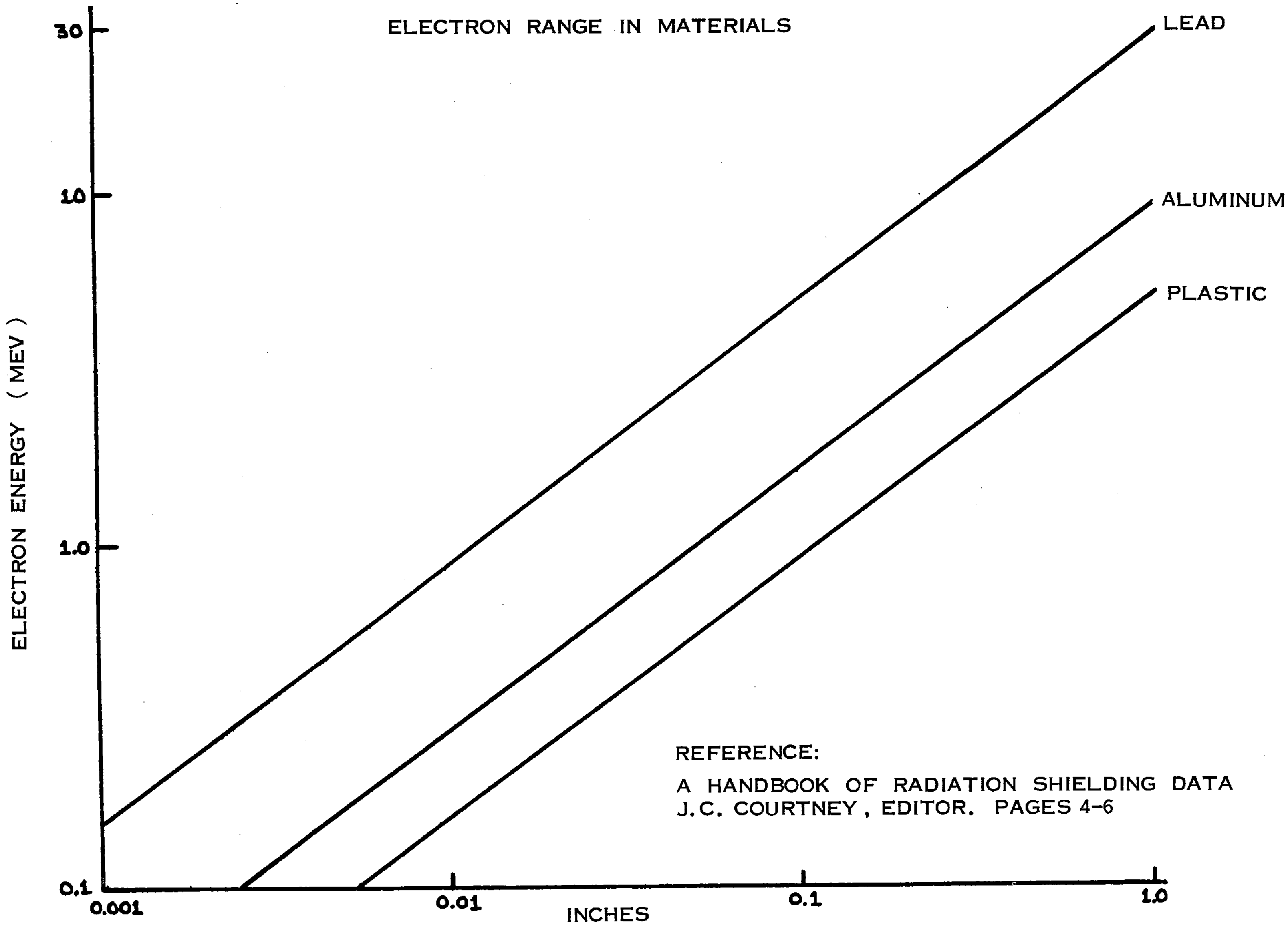
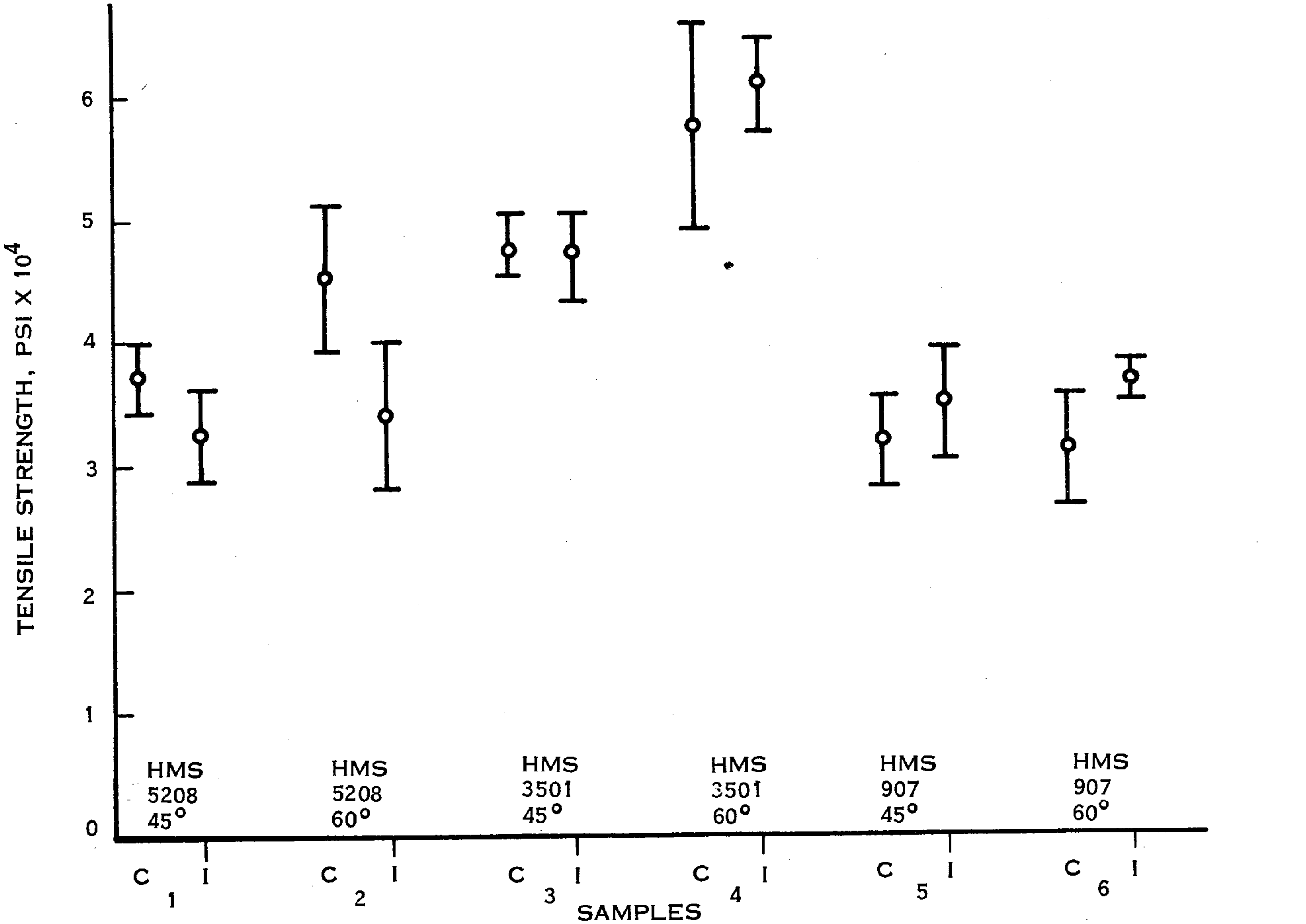

ELECTRON RANGE IN MATERIALS
LEAD
ALUMINUM
PLASTIC
ELECTRON ENERGY (MEV)
30
10
1.0
0.1
0.001
0.01
0.1
1.0
INCHES
REFERENCE:
A HANDBOOK OF RADIATION SHIELDING DATA
J.C. COURTNEY, EDITOR. PAGES 4-6

Figure 9

(Figure 10; Figure 11)

SINCE NON-METALLIC COMPOSITES ARE PRIME CANDIDATES FOR THE FABRICATION OF LSS STRUCTURAL MEMBERS, IT IS IMPORTANT TO ESTABLISH THE ENVIRONMENTAL STABILITY OF THE VARIOUS COMPOSITE FIBER/RESIN SYSTEMS IN ORDER TO ASSURE THE SELECTION OF MATERIALS THAT WILL PROVIDE THE PRESCRIBED LIFETIME IN ORBIT. THIS CHART, AND THE ONE FOLLOWING, PRESENT THE RESULTS OF SOME MSFC IN-HOUSE TESTS THAT WERE MADE USING A 2.5 MEV VAN DE GRAAFF ACCELERATOR. THREE SETS OF COMPOSITE SPECIMENS WERE PREPARED USING THE SAME TYPE OF GRAPHITE FIBER, BUT THREE DIFFERENT EPOXY RESIN SYSTEMS. THE THICKNESS OF THE SPECIMENS RANGED FROM 0.030 TO 0.040 INCH. SOME SPECIMENS WERE USED TO OBTAIN CONTROL PROPERTY DATA, AND THE OTHERS WERE IRRADIATED IN VACUUM WITH 2 MEV ELECTRONS TO A FLUENCE OF 2.6×10^{12} ELECTRONS/cm^2. THIS FLUENCE REPRESENTS THE PREDICTED ELECTRON RADIATION EXPOSURE THAT WOULD BE ENCOUNTERED IN 7 YEARS IN LEO (300 NM 30° INCLINATION) OR IN 7.5 DAYS IN GEO WHICH VIVIDLY DEMONSTRATES THE MORE HOSTILE ENVIRONMENT ASSOCIATED WITH GEO OPERATIONS. THE DATA SHOWS THAT THERE WAS A SIGNIFICANT DEGRADATION IN BOTH THE TENSILE STRENGTH AND THE MODULUS OF THE HMS/5208 FIBER/RESIN SYSTEM. THIS, OF COURSE, REPRESENTS UNACCEPTABLE PERFORMANCE AND ELIMINATES THIS MATERIAL SYSTEM FOR LSS APPLICATION IN EITHER LEO OR GEO. THE PROPERTIES OF THE HMS/3501 AND HMS/907 MATERIAL SYSTEMS WERE IMPROVED BY THIS EXPOSURE. HOWEVER, BEFORE EITHER COULD BE DESIGNATED AS CANDIDATE LSS MATERIALS, TESTS WOULD HAVE TO BE MADE AT HIGHER FLUENCES (TO $\sim 5 \times 10^{15}$ ELECTRONS/cm^2) TO VERIFY STABILITY.

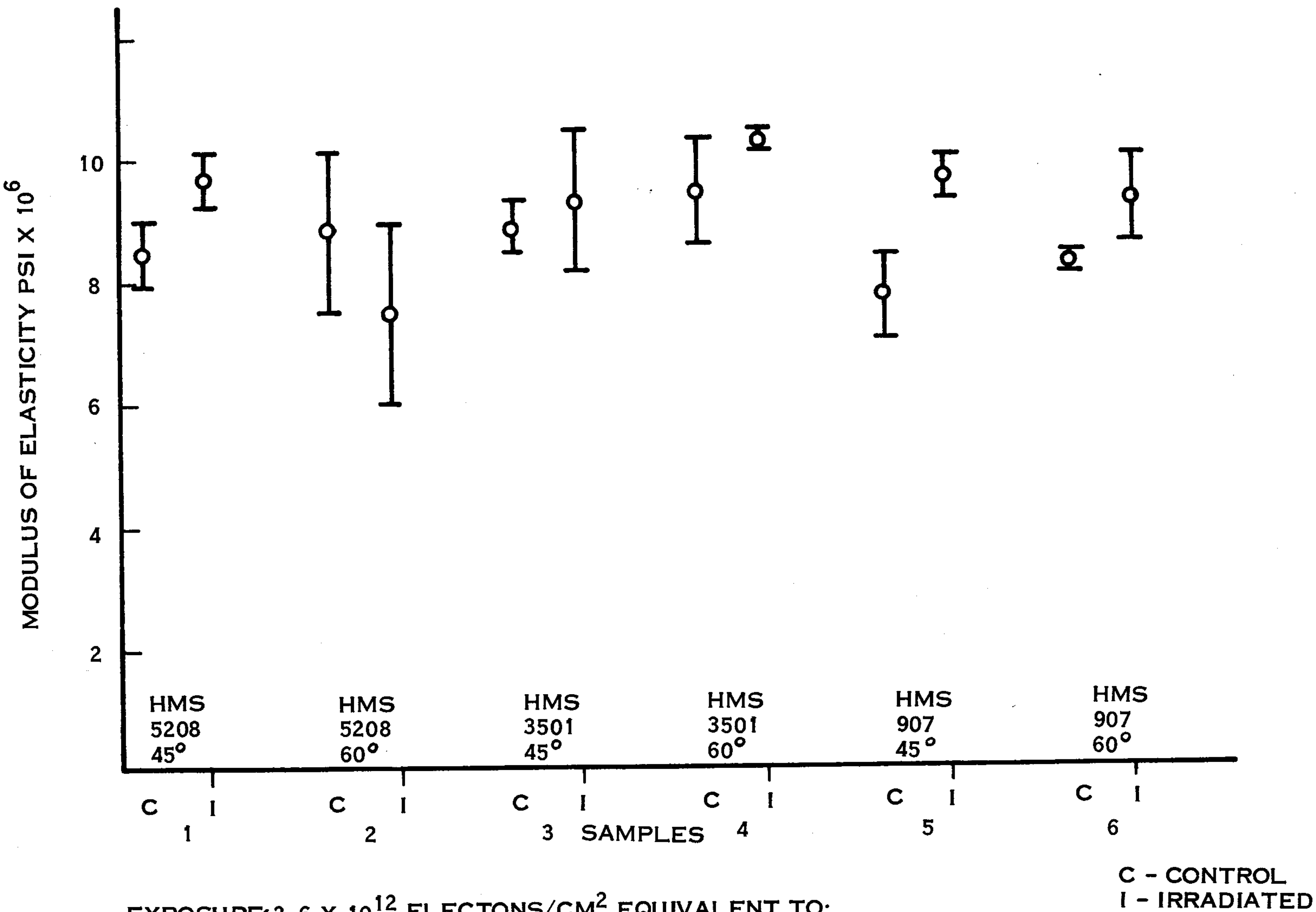
EFFECTS OF RADIATION ON TENSILE STRENGTH



EXPOSURE; 2.6×10^{12} ELECTRONS/CM^2, EQUIVALENT TO:
7 YEARS IN A 300 NAUTICAL MILE 30° ORBIT OR,
7.5 DAYS IN GEOSYNCHRONOUS ORBIT

Figure 10

EFFECTS OF RADIATION ON MODULUS OF ELASTICITY



EXPOSURE; 2.6 X 10^{12} ELECTONS/CM^2, EQUIVALENT TO:
7 YEARS IN A 300 NAUTICAL MILE 30° ORBIT OR,
7.5 DAYS IN GEOSYNCHRONOUS ORBIT

Figure 11

(Figure 12)

THIS CHART SHOWS THE EFFECTS OF THERMALLY CYCLING IN VACUUM FROM -100°F TO +300°F ON THE MECHANICAL PROPERTIES OF THIN GAUGE (.031 INCH) ALUMINUM. AFTER 1000 CYCLES, THE MOST SIGNIFICANT CHANGE WAS A 44% DECREASE IN ELONGATION. THESE TESTS ARE BEING MADE IN-HOUSE AT MSFC TO EVALUATE THE SPACE PERFORMANCE OF THIN ALUMINUM SUCH AS IS BEING CONSIDERED FOR USE AS LSS STRUCTURAL MEMBERS. DATA WILL BE OBTAINED FOR 5000 THERMAL CYCLES FOLLOWED BY ELECTRON IRRADIATION TO 5×10^{15} ELECTRONS/CM^2.

ORGANIZATION:

MARSHALL SPACE FLIGHT CENTER

MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS

NAME:

DATE:

MECHANICAL PROPERTIES OF THERMALLY CYCLED 6061-T6 THIN GAUGE ALUMINUM[1]

CONDITION	UTS (KSI)	YS (KSI)	ELONG. (% IN 2.0 IN.)
UNCOATED	46.9	42.9	12.9
COATED (BAKED)[2]	44.7	41.3	12.1
COATED (BAKED), THERMALLY CYCLED[3]	40.9	38.4	6.8

NOTES:
1. THICKNESS OF AL, 0.03125 INCHES.
2. COATING, KEM LUSTRAL S-65B2 BAKED AT 200°F FOR 4 HOURS.
3. THERMAL CYCLING BETWEEN -100°F AND +300°F FOR 1000 CYCLES AT 1 HOUR PER CYCLE.

Figure 12

ORGANIZATION:

MARSHALL SPACE FLIGHT CENTER

NAME:

DATE:

REFLECTOR MATERIALS TESTS

MATERIAL: 4 MIL ACRYLIC SILVERED POLYESTER

TEST ENVIRONMENT: 10^{-6} TORR, 100 °C, OUTGASSING

RESULTS: MATERIAL SEPARATION AT SILVER/POLYESTER BOUNDARY

MATERIAL: ALUMINIZED 1 MIL MYLAR

TEST ENVIRONMENT: 10^{-6} TORR, PROTON IRRADIATION OF 10^{12} TO 10^{16} PROTONS/CM^2

RESULTS: FAILURE OF MYLAR AT ALL PROTON FLUENCES

MATERIAL: ALUMINIZED 2 MIL KAPTON

TEST ENVIRONMENT: 10^{-6} TORR, PROTON IRRADIATION OF 10^{12} TO 10^{16} PROTONS/CM^2

RESULTS:

PROTONS/CM^2	REFLECTANCE
10^{12}	0.89
10^{13}	0.89
10^{14}	0.89
10^{15}	0.87
10^{16}	0.84
CONTROL	0.87

Figure 13

(Figure 14; Figure 15)

THIS CHART, AND THE ONE FOLLOWING, SUMMARIZE THE MATERIALS AREAS FOR WHICH FURTHER TECHNOLOGY DEVELOPMENT IS NEEDED TO MEET LSS REQUIREMENTS. A PROGRAM ENCOMPASSING THESE AREAS WAS FORMULATED BY A NASA INTERCENTER WORKING GROUP AND SUBMITTED TO THE LSS PROGRAM OFFICE FOR INCORPORATION IN THE LSS PROGRAM PLAN.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS	NAME: DATE:

LSS MATERIALS TECHNOLOGY DEVELOPMENT

- COMPOSITES
  - THERMOPLASTIC AND THERMOSETTING
  - LOW THERMAL EXPANSION/DIMENSIONALLY STABLE
  - RADIATION RESISTANT
  - ULTRA THIN
- METALS
  - THIN GAUGE SHEET
  - WIRE MESH
  - CONDUCTIVE MEMBRANES
- POLYMERS
  - ADHESIVES
  - DIELECTRICS
  - RIGIDIZING
  - THIN FILMS (COATED AND UNCOATED)

Figure 14

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS	NAME: DATE:

LSS MATERIALS TECHNOLOGY DEVELOPMENT (CONTINUED)

- COATINGS
  - REFLECTOR
  - THERMAL CONTROL AND UV PROTECTIVE (INTEGRAL AND APPLIED)
- SPACE ENVIRONMENTAL EFFECTS
  - EVALUATE MATERIALS
    - SCREENING
    - IN-SITU
    - SYNERGISTIC
    - LONG TERM
  - DEVELOP PREDICTIVE MODELS
  - DEFINE FLIGHT EXPERIMENTS
  - PROVIDE DATA BASE FOR DESIGN

Figure 15

(Figure 16)

IN SUMMARY, IT HAS BEEN DEMONSTRATED BY EXAMPLE THAT THE PERFORMANCE AND RELIABILITY OF LARGE SPACE SYSTEMS WILL BE CRITICALLY DEPENDENT ON THE SELECTION OF MATERIALS WHICH WILL EXHIBIT STABILITY IN FUNCTIONAL PROPERTIES IN THE LSS ORBITAL ENVIRONMENT. IN THIS REGARD, THE DEVELOPMENT OF MATERIALS WHICH WILL HAVE A 20 TO 30 YEAR LIFE IS THE MOST SIGNIFICANT PROBLEM FACING THE MATERIALS COMMUNITY. TO PROVIDE THE TECHNOLOGY TO MEET THIS REQUIREMENT REQUIRES AN INTENSIVE EFFORT IN SEVERAL MATERIALS AREAS. THESE AREAS HAVE BEEN DEFINED AND A TECHNOLOGY PLAN PREPARED. TIMELY IMPLEMENTATION OF THIS PLAN IS NECESSARY TO INSURE THE AVAILABILITY OF THE LONG LIFE HIGH PERFORMANCE MATERIALS THAT WILL BE REQUIRED FOR THE LSS PROGRAM.

ORGANIZATION:	MARSHALL SPACE FLIGHT CENTER MATERIALS TECHNOLOGY DEVELOPMENT FOR LONG LIFE LARGE SPACE SYSTEMS	NAME: DATE:

SUMMARY

- LSS PERFORMANCE CRITICALLY DEPENDENT ON MATERIALS SELECTION
- MOST SIGNIFICANT PROBLEM IS 20-30 YEAR LIFE REQUIREMENT
- MATERIALS TECHNOLOGY DEVELOPMENT REQUIRED
  - NEEDS DEFINED
  - PLAN PREPARED
  - TIMELY IMPLEMENTATION NECESSARY

Figure 16

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Materials Technology Development for Long Life Large Space Systems

Approach to Test Planning for Materials

The reusability of the shuttle opens up the potential for satellites with lives beyond the 10 years experienced to date. While needs can be established for exposures ranging from 20 to 30 years, the materials testing effort should be approached as a series of 4 to 5 year programs in which each can be realistically defined, costed and implemented.

PRESENTATION TO GOVERNMENT/INDUSTRY SEMINAR
ON LARGE SPACE SYSTEMS TECHNOLOGY (LSST),
LANGLEY RESEARCH CENTER

LARGE STRUCTURE CONTROL DEVELOPMENT CONCEPTS

G. RODRIGUEZ

Jet Propulsion Laboratory
January 18, 1978

This paper presents viewpoints on large structure control evolving from the solar sail study conducted recently at JPL. The objective is to make optimum use of insights gained in the study in order to assess required large structure control developments. While the Halley's comet rendezvous mission for which the sail was under prime consideration is no longer in NASA's plans, the sail is an ideal reference configuration to identify control development needs as it may have been the first large structure to be considered seriously for a near-future NASA mission.

OVERVIEW (Figure 1)

In the paper, the major sail control challenges are identified as configuration development, size and flexibility, and system uncertainty. These challenges are illustrated with the sail control design evolution. Distributed and adaptive control are identified as two major conceptual areas requiring development in order to fly the sail and more general large space structures.

OVERVIEW

- MAJOR CHALLENGES
- FUNDAMENTAL TRADEOFFS
- SOLAR SAIL CONCEPTS
- SAIL CONTROL DESIGN PROBLEMS
- REQUIRED CONTROL DEVELOPMENTS
- CONCLUSIONS

Figure 1

GR-2
1/18/78

SOLAR SAIL LARGE STRUCTURE CONTROL (Figure 2)

The single major sail control challenge was the development of a structure/control integrated configuration. Fundamental to configuration development were: the selection between spinning and nonspinning concepts, the definition of control mechanizations (mass expulsion, solar pressure, etc.), the design of vehicle shape to minimize disturbances including those of the solar pressure itself, and the need to provide failure protection. Of course, these considerations are common to most spacecraft designs. The unique challenge of the sail was that the control mechanization had to be an integral part of the structure not just an attachment as a gas jet would be in many current spacecraft. This challenge created a need for close working interfaces among a variety of technical disciplines such as structures, control, thermal control, etc. This point can be illustrated with the sail design evolution as discussed in the sequel.

SOLAR SAIL LARGE STRUCTURE CONTROL

- SINGLE MAJOR CHALLENGE
  - STRUCTURE/CONTROL INTEGRATED DESIGN
- FUNDAMENTAL TRADEOFFS
  - SPINNING VS NONSPINNING
  - CONTROL MECHANIZATION OPTIONS
  - ENVIRONMENTAL DISTURBANCES
  - FAILURE SENSITIVITY

Figure 2

GR-3
1/18/78

NONSPINNING SAIL (Figure 3)

A solar sail vehicle would have as one of its main components a large, lightweight reflective sail. Propulsive force within the solar system would be derived from sunlight reflection from the sail. Photons striking the sail would be reflected back, a change in momentum is experienced which would be expressed as a force acting on the sail. Two fundamental sail concepts were under study: nonspinning and spinning. The nonspinning sail is a three-axis stabilized vehicle formed by a sail module and a support structure. The sail has four sides and is roughly pyramidal in shape. Each side is 800 m in length, and the sail apex is 63 meters above its base.

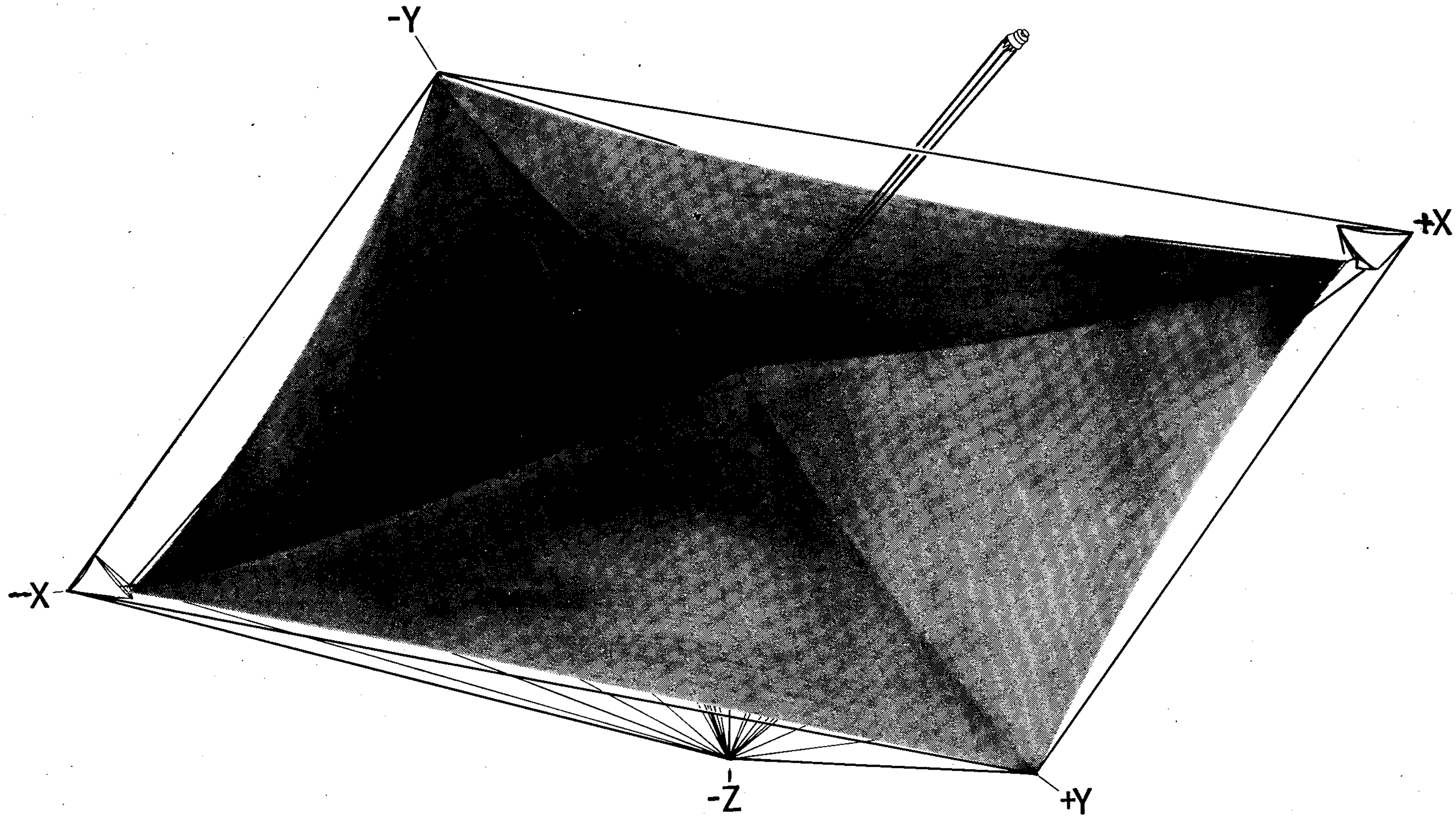


Figure 3

SPINNING SAIL (Figure 4)

The spinning sail is a long multiblade spinner based on the heliogyro concept developed by R. H. MacNeall and J. M. Hedgepeth at Astro Research Corp. in the late sixties and studied further by them and JPL in the recent studies. The heliogyro dynamics and control are very similar to those of a helicopter with control being achieved by pitching or rotation of the blades about their long axis. The nonspinning and spinning configurations were developed in parallel with the common objective among all sail team members to optimize both configurations. At a given point in time, a selection between the two was made and the heliogyro was closer to satisfying the overall mission objectives. However, the remainder of the paper concentrates on the nonspinning concept with which the author was involved primarily.

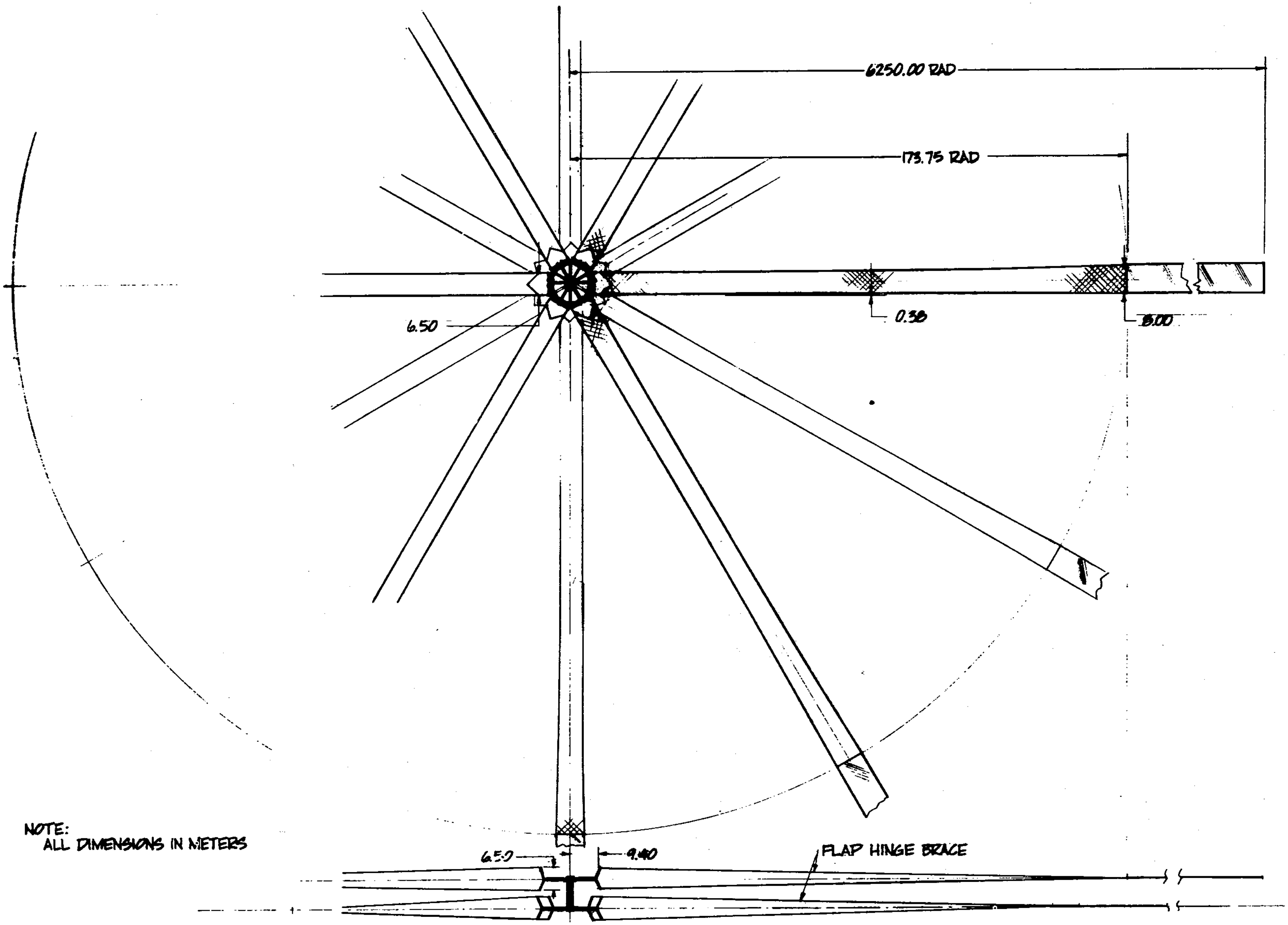

6250.00 RAD
173.75 RAD
6.50
0.38
15.00
6.50
9.40
FLAP HINGE BRACE
NOTE:
ALL DIMENSIONS IN METERS

Figure 4

NONSPINNING SAIL CONTROL CONCEPTS (Figure 5)

Many candidate concepts were studied for control of the nonspinning sail. The vane, sheet and ballast mass concepts illustrate best the sail control design evolution.

NON-SPINNING SAIL CONTROL CONCEPTS

- VANE
- SHEET
- BALLAST MASS

Figure 5

GR-6
1/18/78

VANE CONTROL CONCEPT (Figure 6)

The vane attitude control option uses four single-degree-of-freedom vanes to supply three-axis control. The gimbal degree of freedom for vane i (V_i) is γ_i. Pitch (x) control can be obtained by rotating V_3 through 90 deg or less, shifting the center of solar pressure along -y, and thus producing a torque about x. Similarly, yaw (y) torque is generated by rotating V_2. Roll (z) control is produced by differential rotation of V_1 and V_3, or V_2 and V_4, or both sets simultaneously. This concept was suggested by a Battelle Columbus Laboratories 1973 study and had intuitive appeal at JPL because of its similarities with thrust vector control mechanizations in recent Mariner spacecraft.

CONTROL CONCEPTS
FOUR SINGLE DEGREE OF FREEDOM VANES OPTION

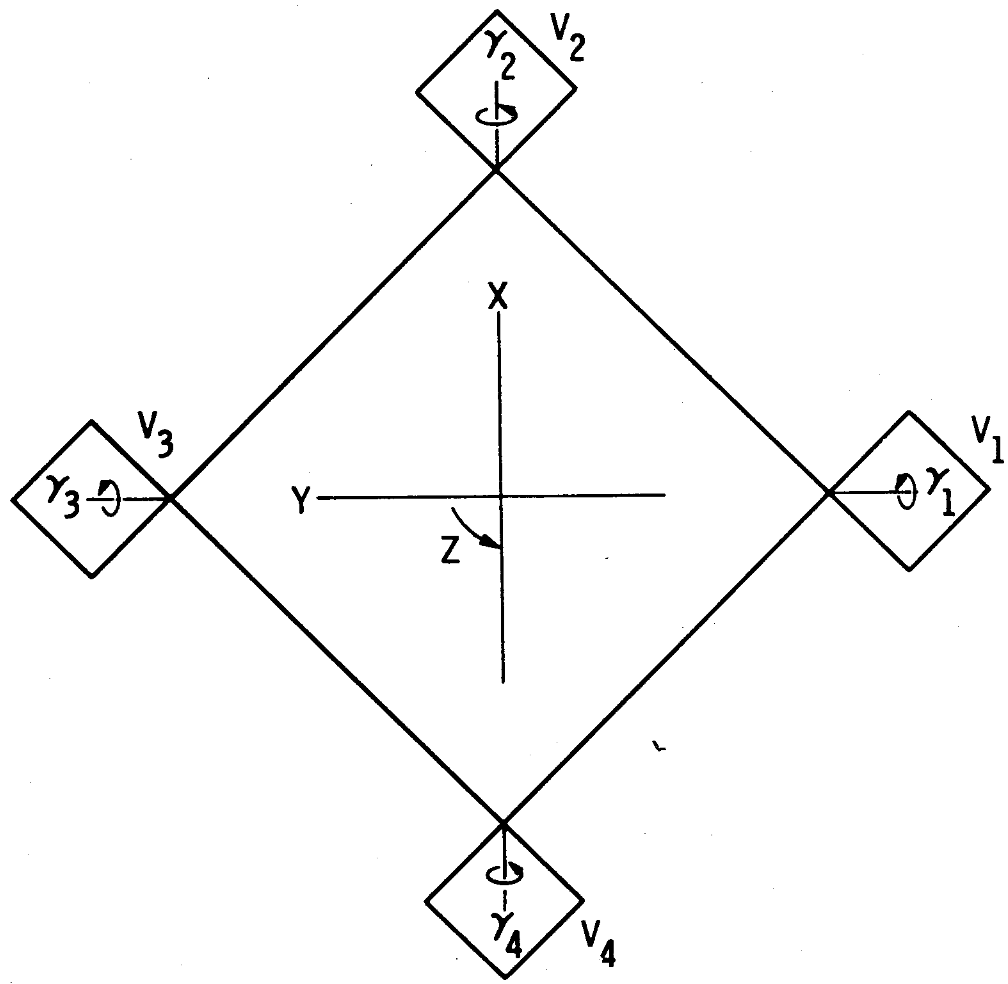


Figure 6

VANE CONTROL CONCERNS (Figure 7)

A number of limitations in the concept were, however, identified such as: large vane size (200 m on each side) required to offset huge (500 n-m) disturbance torques caused by solar pressure acting on the sail surface; packaging and deployment problems caused by large vane size; failure sensitivity, as a single vane failure implied loss of three-axis control capability; structure/control interaction, as vane rotation disturbed the main sail structure; and inaccurate control torque knowledge due to unpredictable effects such as sail deformations, nonhomogeneous sail reflectivity changes, mass and pressure center shifts, sail holes and tears, etc. For these and other reasons, the design proceeded to the sail translation concept.

VANE CONTROL CONCERNS

- LARGE VANE SIZE (200 m)
- PACKAGING AND DEPLOYMENT
- FAILURE SENSITIVITY
- STRUCTURE/CONTROL INTERACTION
- UNPREDICTABLE CONTROL TORQUES

Figure 7

GR-8
1/18178

SAIL TRANSLATION CONCEPT (Figure 8)

The sail translation scheme utilizes only two vanes for roll control as described above. However, pitch and yaw control are produced by translation of the sheet material parallel to the pitch and yaw axes. This is accomplished with outhaul winch actuators at the sail corners together with inhaul winch actuators at the sail center. With the translation of vehicle mass and pressure centers, pitch and yaw torques are generated. This concept was considered attractive at first because it did not require large vanes and reduced vehicle weight by using winch actuators for the dual purpose of attitude and shape control.

CONTROL CONCEPTS
SINGLE SAIL TRANSLATION, ROLL VANES

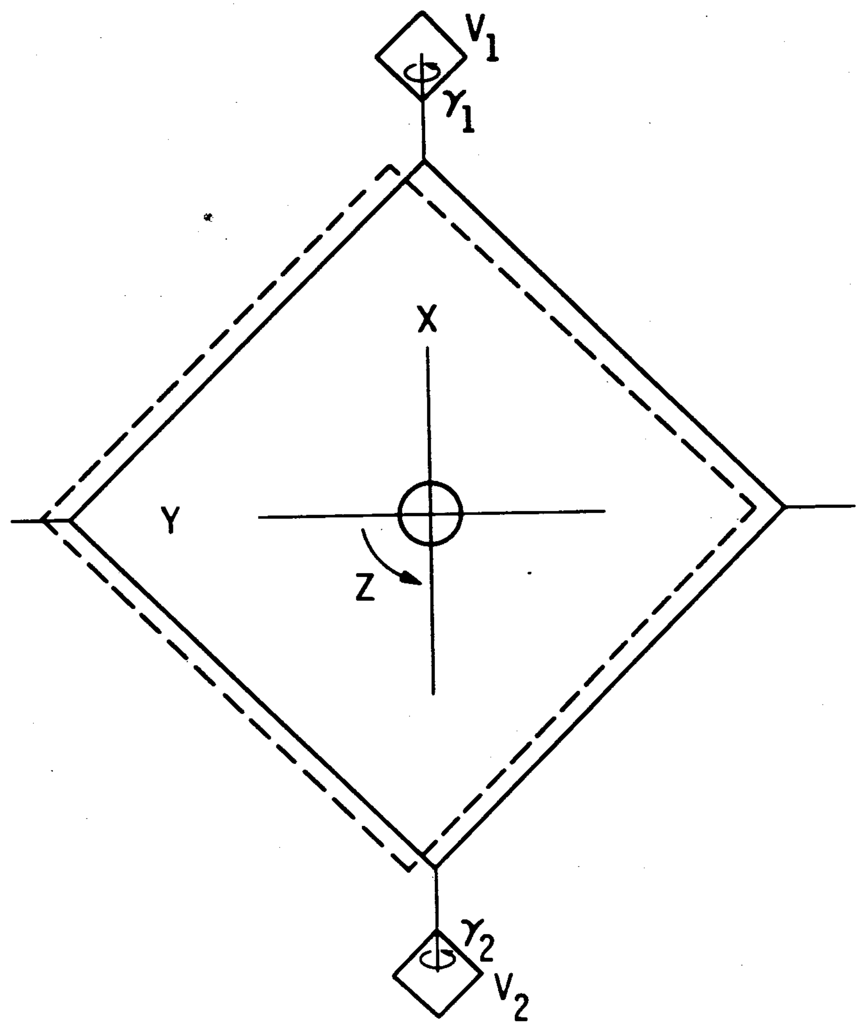


Figure 8

SAIL TRANSLATION CONCERNS (Figure 9)

Sail translation concept concerns included: the need to move half the vehicle mass (4000 kg) to accomplish the translation maneuver, sail wear due to sail distortion, failure sensitivity, structure/control interactions, complicated dynamical modeling, and inaccurate control torque knowledge as sail translation implied unpredictable sail distortions and solar pressure disturbance torques. The design proceeded to the ballast mass idea.

SAIL TRANSLATION CONCERNS

- LARGE MASS (2000 kg) MOTION
- SAIL WEAR DUE TO SAIL DISTORTION
- FAILURE SENSITIVITY
- STRUCTURE/CONTROL INTERACTIONS
- COMPLICATED DYNAMICAL MODELING
- UNPREDICTABLE CONTROL TORQUES

Figure 9

GR-10
1/18/78

BALLAST MASS CONCEPT (Figure 10)

In the ballast mass configuration, roll control vanes are still present. However, control for the other two axes is accomplished by pure mass center control. This is provided by suspending a ballast mass by cables from the four sail corners. The mass is located on the Sun side of the vehicle. By controlling the lengths of the cables, the vehicle mass center can be shifted in the x,y plane. Noncoincidence of the mass and pressure centers generates pitch and yaw torques.

CONTROL CONCEPTS
BALLAST MASS, VANES OPTION

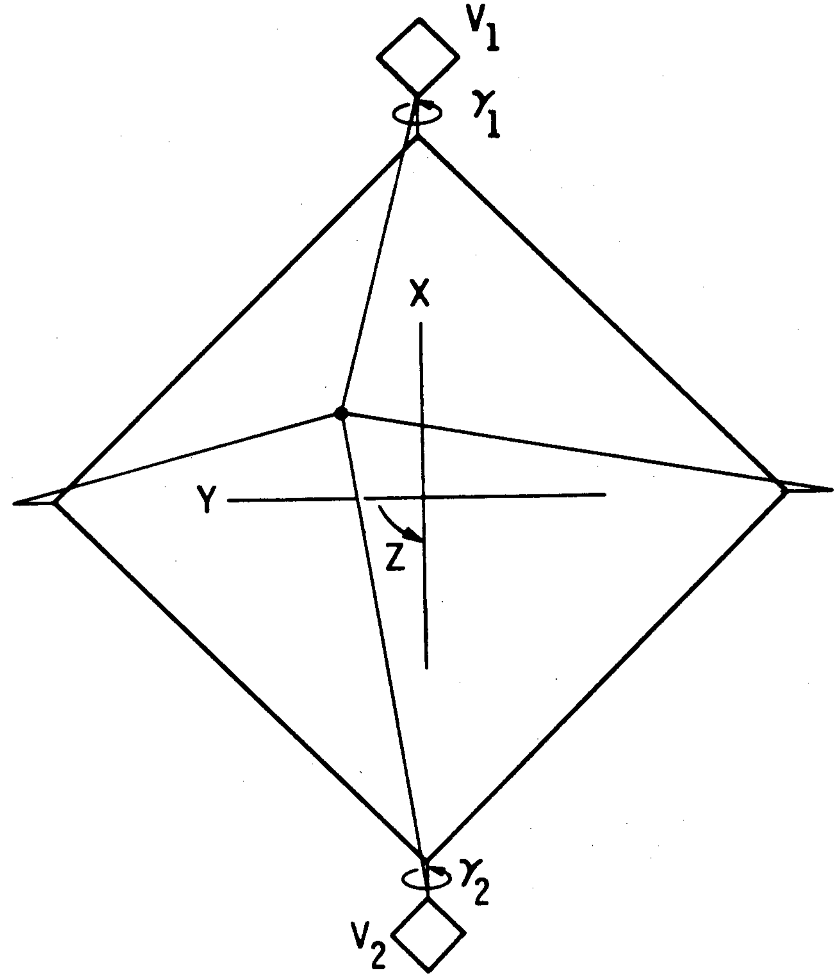


Figure 10

BALLAST MASS CONSIDERATIONS (Figure 11)

The ballast mass idea has some positive features such as its reliable mechanization by cables instead of actuators and its ability to generate large control torques with relatively small masses (100-200 kg) in spite of possible major failures in the sail sheet. There were deployment concerns because the ballast mass had to be deployed after the sail structure deployment. Inaccurate knowledge of control torques was a problem common to most of configurations studied. Although several other square sail configurations were considered, the two-vanes ballast idea was selected as the final nonspinning baseline concept in the JPL studies. Significant efforts remained to improve this design in order to overcome the many control challenges summarized in the sequel.

BALLAST MASS CONSIDERATIONS

- RELIABLE MECHANIZATION
- LARGE CONTROL TORQUES
- DEPLOYMENT CONCERNS
- UNPREDICTABLE CONTROL TORQUES

Figure 11

GR-12
1/18178

SUMMARY OF SAIL CONTROL DESIGN PROBLEMS (Figure 12)

In addition to configuration development, the sail control design problems were due to vehicle flexibility and system uncertainty. Vehicle flexibility had the potential for degradation of attitude/shape control performance, implied complicated mechanization and modeling interfaces (structure, control, thermal, disturbance, etc.), and required that the controller design account for low natural frequencies within its bandwidth (0.01 - 0.001 Hz) and for system uncertainty due to inaccurate dynamic response characterization and vehicle shape knowledge. On-board thrust vector control and a high-precision low-thrust accelerometer may have been required to reduce resultant thrust vector errors.

SUMMARY OF SAIL CONTROL DESIGN PROBLEMS

- CONFIGURATION STRUCTURE/CONTROL INTEGRATED DESIGN
- VEHICLE FLEXIBILITY
  - POTENTIAL PERFORMANCE DEGRADATION
  - COMPLICATED INTERFACES
    - STRUCTURE/CONTROL
    - THERMAL
    - DISTURBANCE ENVIRONMENT
    - MODELING
  - ANALYTICAL CONTROLLER DESIGN
    - LOW NATURAL FREQUENCIES
    - SYSTEM UNCERTAINTIES

Figure 12

GR-13
1/18/78

REQUIRED CONTROL DEVELOPMENTS (Figure 13)

Two major conceptual areas can be identified where advances are required in order to solve problems due to size and flexibility and system uncertainty. Distributed control where the control system mechanization (actuators, sensors, etc.) are mounted throughout the structure could provide: active shape and vibration control, a potential reduction in vehicle weight, and inherent redundancy and fault tolerance. Adaptive control where the control system monitors and adjusts its own performance could provide: vehicle autonomy, a trend toward performance optimization, and fault tolerance/correction including automatic transfer of the vehicle to a safe operational state in case of major failures (e.g. a sail tear). Vehicle autonomy also implies nondedicated mission operations. There is agreement among the NASA centers participating in the LSST program on the need for developments in these two areas. The following outline presents a view of what may be involved in these developments.

SUMMARY OF SAIL CONTROL DESIGN PROBLEMS

- SYSTEM UNCERTAINTY
  - VEHICLE DYNAMICS RESPONSE CHARACTERIZATION
    - CONTROL TORQUES
    - STRUCTURAL FREQUENCIES/MODE SHAPES
    - DAMPING
  - SAIL SHAPE KNOWLEDGE
    - DISTURBANCE TORQUES
    - RESULTANT THRUST VECTOR

Figure 13

GR-14
1/18/78

DISTRIBUTED CONTROL DEVELOPMENT CONSIDERATIONS (Figure 14)

Advances are required in the areas of mechanization definition, modeling, analytical controller design and flight performance verification. Structure, control, disturbance, etc. models with clean interfaces are required for pre-flight analysis and controller design. Models are not end items but are intended to provide optimum support to the large structure control developments. Analytical controller design includes control hardware placement, reduced-order controller design (due to the large number of degrees-of-freedom characterizing a large structure), and adaptive control. Flight data analysis is required for in-flight dynamic response evaluation, control system and vehicle calibrations, and overall reduction in system uncertainty.

REQUIRED CONTROL DEVELOPMENTS

- DISTRIBUTED CONTROL
  - SHAPE, VIBRATION CONTROL
  - REDUCE VEHICLE WEIGHT
  - REDUCE ACTUATOR SIZE
  - REDUNDANCY / FAULT TOLERANCE
- ADAPTIVE CONTROL
  - VEHICLE AUTONOMY
  - PERFORMANCE OPTIMIZATION
  - FAULT TOLERANCE/ CORRECTION
  - NONDEDICATED MISSION OPERATIONS

Figure 14

SUMMARY (Figure 15; Figure 16)

The single major sail control challenge was configuration development, as illustrated by its design evolution. Other challenges are due to size and flexibility and system uncertainty. Advances are required in the areas of distributed and adaptive control for active attitude/shape control and to provide vehicle autonomy and reduce system uncertainty. Although distributed and adaptive control may not necessarily be required in all cases, these two areas have the best potential for providing a systematic solution to the challenges of large structure control.

DISTRIBUTED CONTROL DEVELOPMENTS

- MECHANIZATION
  - SURFACE SENSING SYSTEMS
  - ACTUATING SYSTEMS
  - COMPUTER NETWORKS
- MODELING
  - STRUCTURE, CONTROL, DISTURBANCE, ETC. PRE-FLIGHT DYNAMIC MODELING
  - MODEL ORDER REDUCTION FOR CONTROL DESIGN
- ANALYTICAL CONTROL DESIGN
  - CONTROL HARDWARE (SENSOR, ACTUATOR, ETC.) PLACEMENT
  - REDUCED-ORDER CONTROLLER DESIGN
  - ADAPTIVE CONTROL
- IN-FLIGHT PERFORMANCE VERIFICATION
  - FLIGHT DATA PROCESSING
  - ANALYTICAL MODEL VERIFICATION

Figure 15

SUMMARY

SINGLE MAJOR CHALLENGE: CONFIGURATION DEVELOPMENT

PROBLEMS:

- SIZE AND FLEXIBILITY
- SYSTEM UNCERTAINTY

ADVANCES:

- DISTRIBUTED CONTROL
- ADAPTIVE CONTROL

Figure 16

CONTROL CONCEPTS FOR LARGE SPACE STRUCTURES

R. C. QUARTARARO

ROCKWELL INTERNATIONAL

JANUARY, 1978

CONTROL CONCEPTS FOR LARGE SPACE SYSTEMS

INTRODUCTION (Chart 1)

The large space systems planned for the next decade and beyond present a major challenge to control engineers. Many control technology advancements will be required to satisfy new requirements.

A comprehensive program to develop the required control technology has been started at Rockwell International's Space Division. A few of the concepts under consideration for attitude, figure, and vibration control of large, flexible space systems will be highlighted in this presentation. In addition, an overview of the Space Division's IR&D program will be presented. The direction of the IR&D program has been influenced by requirements for electro-optical systems, Shuttle erectable structures and Satellite Power Stations (SPS).

CHART 1

CONTROL CONCEPTS FOR LARGE SPACE SYSTEMS

- ADVANCEMENTS REQUIRED FOR CONTROL OF LARGE SPACE SYSTEMS
- CONCEPTS UNDER DEVELOPMENT AT ROCKWELL
  - ATTITUDE CONTROL
  - FIGURE CONTROL
  - VIBRATION CONTROL
- IR&D PROGRAM HAS BEEN DRIVEN BY
  - ELECTRO-OPTICAL SYSTEMS (5-50 METERS)
  - SHUTTLE ERECTABLE STRUCTURES (100-1,000 METERS)
  - SPACE POWER SYSTEMS (1-30 KILOMETERS)

NEW CHALLENGES (Chart 2)

The need for control system advancements stems from the many new challenges resulting from the greater size and new performance requirements for large space systems. In order to accommodate the many new aspects of large space systems, advancements are required in all areas of control technology. New control concepts are needed to cope with the problems of all-flexible systems; upgraded analytical techniques and computer programs are needed for the more demanding problems of synthesizing a control system for spacecraft with distributed actuators and uncertain dynamics, and systems which must perform attitude, vibration and figure control simultaneously; new sensors and actuators are needed to provide much larger control forces and to deal with the new problems of structural control; improved control electronics with greater capacity and longer life are required; and laboratory and flight experiments will be needed to verify newly developed concepts and hardware and to provide a data base to aid designers of operational systems.

NEW CHALLENGES

NEW CONTROL REQUIREMENTS

INCREASED SIZE $\Longrightarrow$

- GREATER ENVIRONMENTAL EFFECTS
- LARGER CONTROL TORQUES AND FORCES ✓
- MORE FLEXIBLE ✓
- NO FULL-SCALE GROUND TESTS
- SPACE ASSEMBLY

NEW PERFORMANCE REQUIREMENTS $\Longrightarrow$

- ACTIVE FIGURE CONTROL ✓
- RAPID MANEUVERS OF LARGE SYSTEMS ✓
- ORBIT TRANSFER OF LARGE SYSTEMS
- LONGER LIFE

ADVANCEMENTS NEEDED

- CONTROL CONCEPTS ✓
- ANALYTICAL TECHNIQUES, COMPUTER PROGRAMS
- SENSORS, ACTUATORS
- ELECTRONICS
- LAB AND FLIGHT EXPERIMENTS

CONTROL CONCEPTS (Charts 3 - 10)

Control concepts are required to deal with requirements of varying degrees of difficulty, as can be illustrated by results from a recent study of Shuttle erectable structures performed by Rockwell for LaRC.* Chart 3 pictures a high concentration ratio solar array which could be erected from the Shuttle by using struts of practical minimum gauge and lengths compatible with the Shuttle bay to form the structural truss work. Such a system exhibits a relatively high fundamental vibration frequency for sizes up to 1 kilometer by 1 kilometer. If a 1200m x 1200m array, or smaller, is in low Earth orbit and required to point towards the Sun with 1° accuracy, it will "appear" to the attitude control system to be rigid. That is, the control system will not excite significant vibrational motions, and there will be no possibility of unstable interactions between the structure and the attitude control system. The attitude of solar arrays of this class can be controlled by conventional techniques with scaled-up actuators.

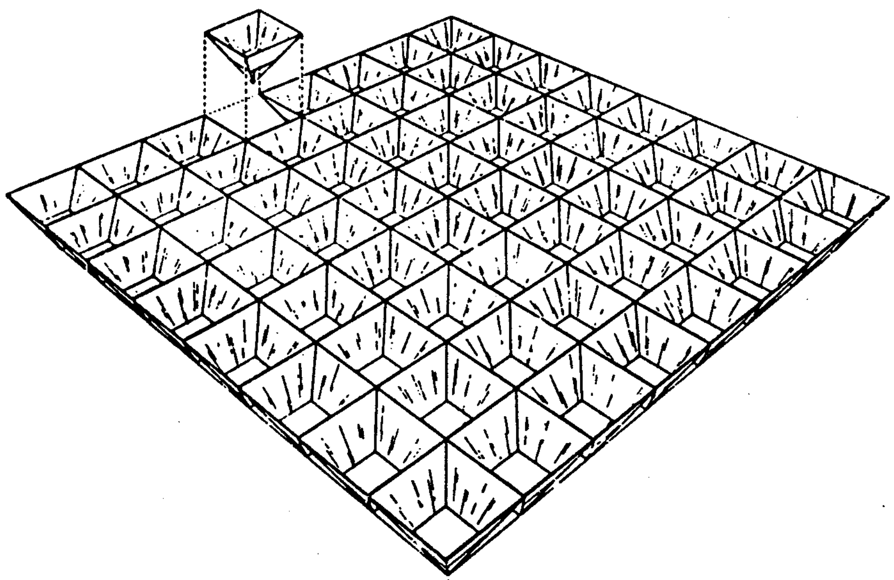
Structural vibrations can couple with the control system and they can be excited by oscillating gravity gradient torques if the array is larger than 1200m x 1200m. In general, this larger class of vehicles requires a structural vibration control system. The complexity of the control system increases as the number of vibration modes in the <0.04 Hz range increases. New concepts are required for all-flexible systems in which many vibration modes must be controlled.

A new class of control problems is also introduced if there is an additional requirement to accurately maintain a prescribed structural figure (shape).

*Reference: NASA CR-145206, "Advanced Technology Laboratory Program for Large Space Structures, "Parts 1 & 2, Final Report," Revised May 1977

CHART 3

CONTROL OF SHUTTLE ERECTABLE STRUCTURES



HIGH CONCENTRATION RATIO SOLAR ARRAY

- L x L x 14.26 M
- PENTAHEDRAL TRUSS
- 1° POINTING
- LOW EARTH ORBIT
- GRAVITY GRADIENT

FUNDAMENTAL VIBRATION FREQUENCY

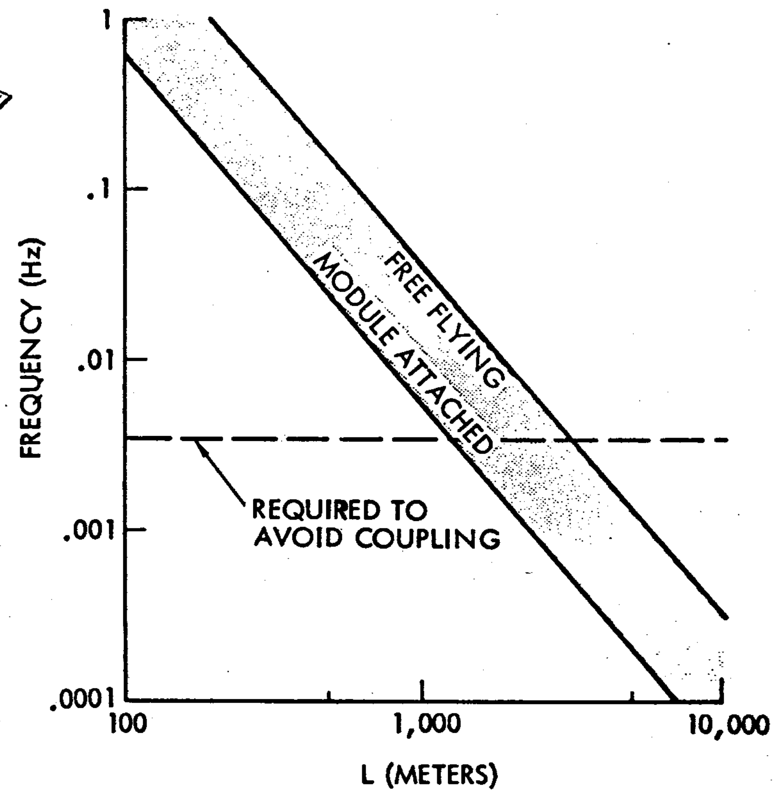
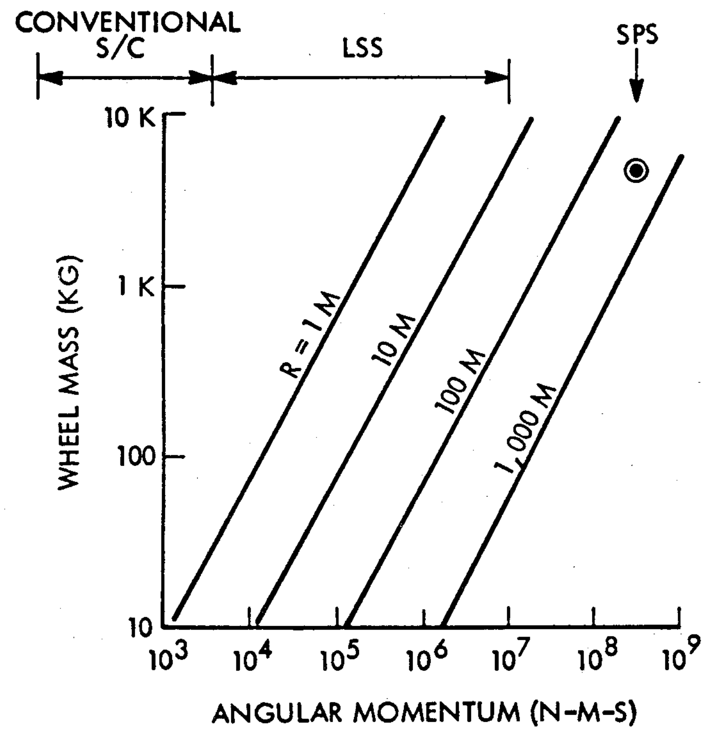
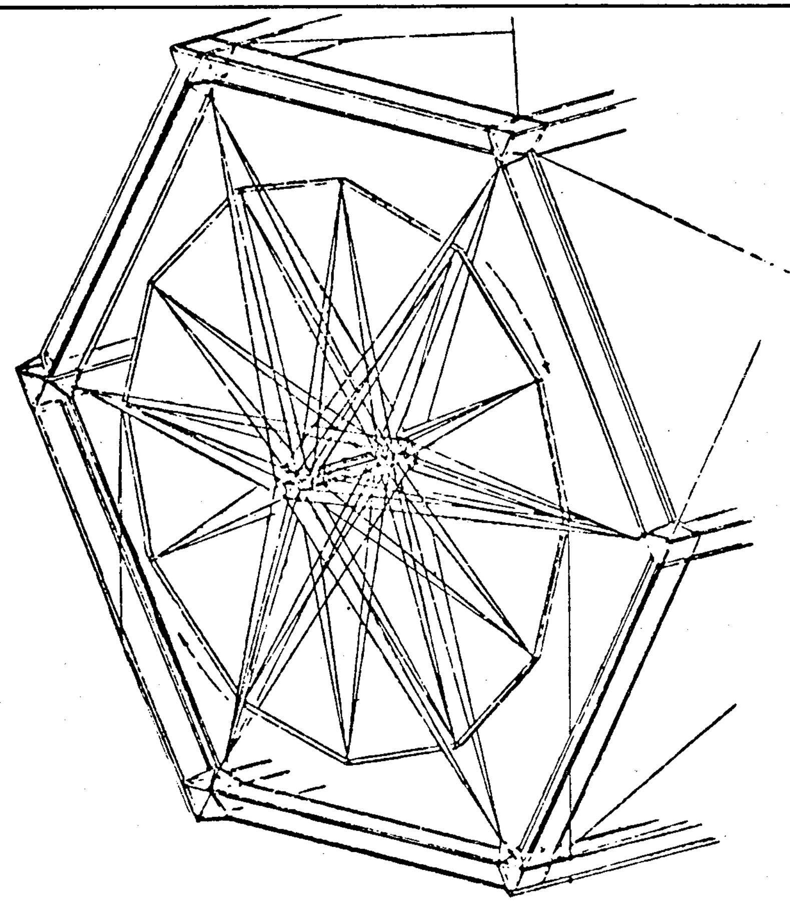


Chart 4 addresses the application of momentum-storage attitude-control techniques to systems for which scaled-up conventional techniques are applicable. The parametric plot of wheel mass versus angular momentum storage requirement for various wheel radii indicates that there is a great saving in wheel mass if the wheel's radius is increased. For example, a large space system which must store 10^5 n-m-s will require several wheels having a total mass approximately 1,000 kilograms if the wheel radii are 1 meter; but a 100 kilogram wheel will suffice if a 10 m radius is selected. A point design for an SPS wheel is pictured in the right half of Chart 4.

The more demanding challenges of vibration and figure control of a large, flexible space structure have been addressed at Rockwell. Much of the effort has dealt with the application modal control techniques. With this method, all (attitude, vibration and figure) small displacements of the vehicle are represented as a summation of modal vectors (shape functions). One special method for controlling modal displacements - independent mode control - has been applied in large space system studies. The concept involves the simultaneous use of several actuators to force displacements in one mode without disturbing some other modes. The concept is illustrated by comparing it with a more "conventional" approach.

MOMENTUM WHEELS FOR ATTITUDE CONTROL





SPS DESIGN EXAMPLE

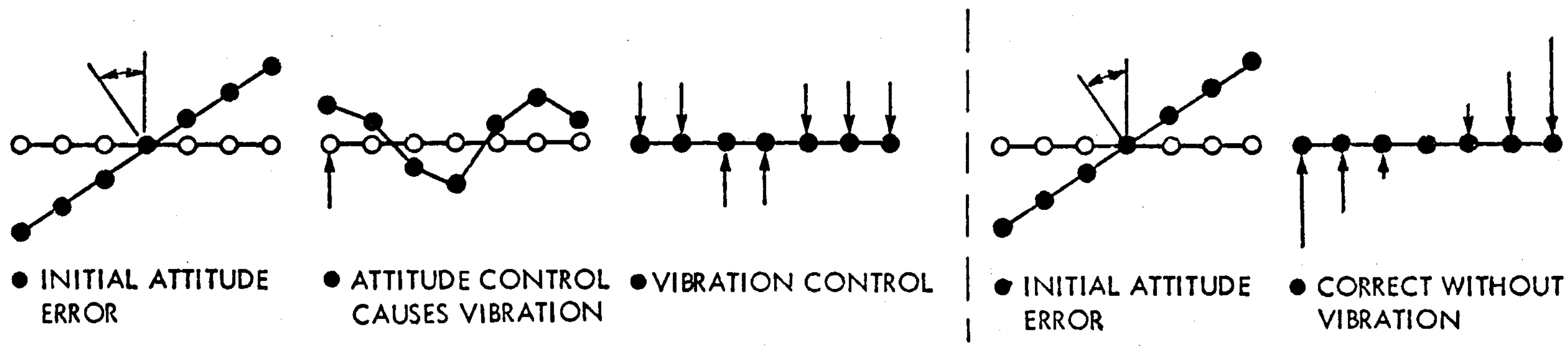
- RADIUS: 350 M
- MAX SPEED: 6.1 RPM
- MTL: ALUMINUM
- FREQ: 0.22 Hz

INDEPENDENT MODE CONTROL

The schematic sequence in the lower left portion of Chart 5 illustrates a method for correcting the attitude of a flexible space system. Jet firings made to correct an attitude error are shown to excite vibrational motions which are then settled by an active or passive vibration control scheme. In contrast, an independent mode controller corrects an attitude error by firing several jets (or combinations of jets and structural actuators) with the relative ratio of forces selected to cause attitude motions without exciting low frequency vibrational motions. Thus, the attitude error can be corrected without inducing unwanted vibration oscillations. This control concept is also applicable to figure control.

INDEPENDENT MODE CONTROL

- <u>APPLICABLE</u> TO ATTITUDE, FIGURE AND VIBRATION CONTROL
- <u>MODAL REPRESENTATION</u>: DISPLACEMENT - SUMMATION OF SHAPE FUNCTIONS (MODES)
- <u>INDEPENDENT MODE CONTROL</u>
  - FORCE ONE MODE WITHOUT FORCING OTHERS
  - SIMPLIFIES CONTROL SYNTHESIS
  - ELIMINATES SOME INTERACTIONS
  - INSIGHT



CONTROL CONCEPTS, MODAL CONTROL

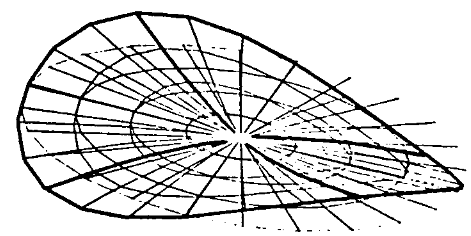
The next two charts (Charts 6 and 7) illustrate an application of independent mode control to a 300 meter, Shuttle-erectable platform. The vehicle modes (Chart 6) include six rigid body modes to represent translational and rotational motions with no structural distortion. The first nine modes (six rigid body plus three vibration) can be controlled independently by nine or more actuators. For illustrative purposes, nine thrusters are employed. (In practice, some structural actuators would be used).

CONTROL CONCEPTS
MODAL CONTROL EXAMPLE

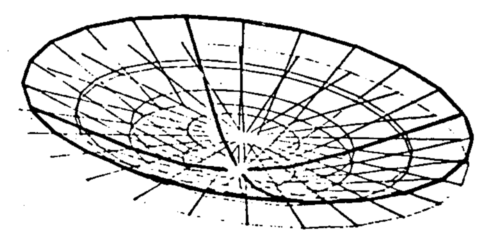
MODES FOR A 300 METER PLATFORM

453590 kg (1,000,000 lb) MISSION EQUIPMENT
22680 kg (50,000 lb) STRUCTURE

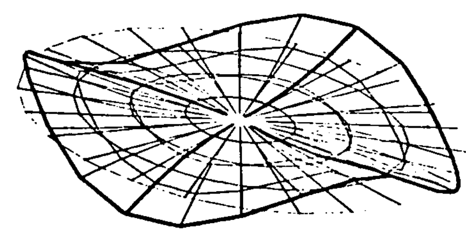
MODE NUMBER	FREQUENCY (Hz)
1-6	0
7	0.191
8	0.191
9	0.329
10	0.447
11	0.447
12	0.753
13	0.753
14	0.795
15	0.795
16	1.24
17	1.24



$f_7 = 0.191$ Hz



$f_9 = 0.329$ Hz

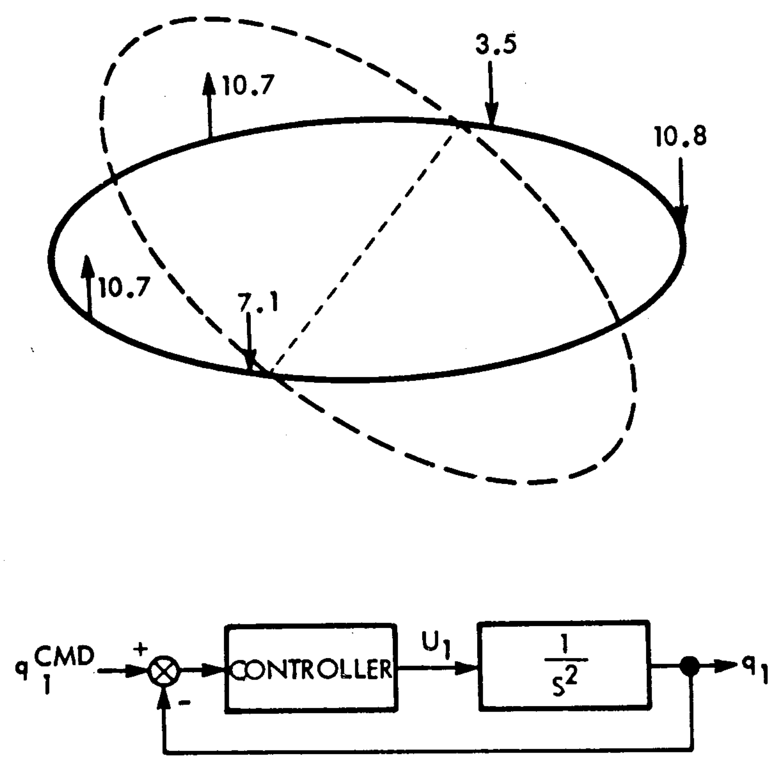


$f_{10} = 0.447$ Hz

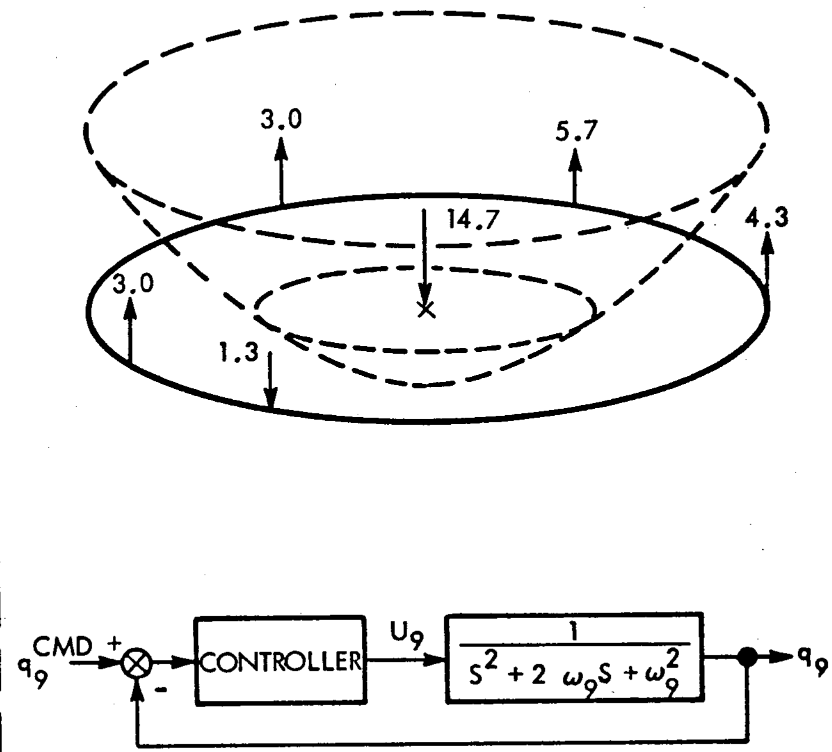
Control of two of the modes is illustrated in Chart 7. The nonzero forces which must be applied are indicated. The control forces must be applied so as to preserve the ratio of forces illustrated. The absolute force level is determined by the control system, and each mode is controlled by a single, independent control loop.

CONTROL CONCEPTS
MODAL CONTROL EXAMPLE (CONTD)

INDEPENDENT MODE CONTROL



RIGID BODY (POINTING) MODE



3rd VIBRATIONAL MODE

MODE CONTROL CONSIDERATIONS

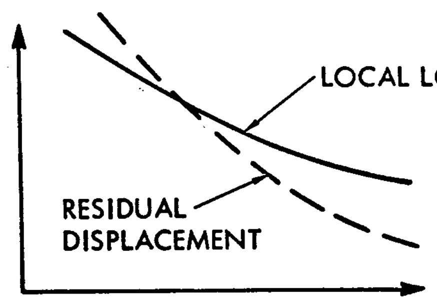
Important considerations in designing a mode controller (independent or otherwise) are listed in Chart 8. The number and placement of actuators must be carefully selected in order to limit the force required to displace the controlled modes while simultaneously limiting the displacements in the uncontrolled modes. The types of actuators selected may also have a significant affect on performance. As the number of modes to be controlled increases, the system's dynamic behavior (at least as determined by unconfirmed analyses) becomes more uncertain and adds to the difficulty in locating actuators and establishing force ratios.

MODE CONTROL CONSIDERATIONS

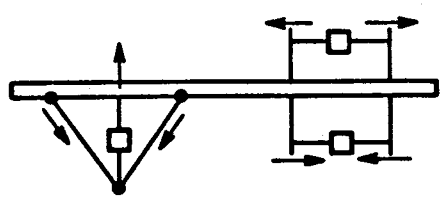
- PLACEMENT OF ACTUATORS

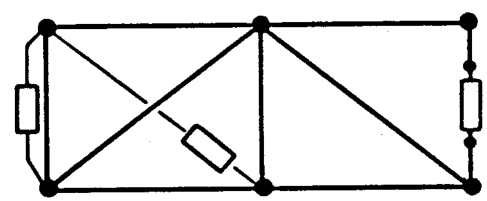
LOCATION CRITERIA: MINIMIZE $\dfrac{\text{(DISPLACEMENT IN UNCONTROLLED MODES)}}{\text{(DISPLACEMENT IN CONTROLLED MODES)}}$

- NUMBER OF ACTUATORS



- TYPES OF ACTUATORS





- MODEL UNCERTAINTIES

UNCERTAINTY INCREASES AS NUMBER OF MODES TO BE CONTROLLED INCREASES

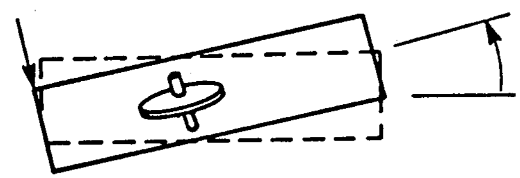
TWO-BODY CONTROL

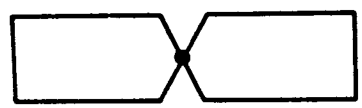
Another concept under development at the Space Division is two-body control (Chart 9), which provides advantages for some large systems which must reorient rapidly. It is accomplished by dividing a vehicle into two parts, the pointed and the reaction bodies, connected by a two-axis gimbal. The pointed body is reoriented by activating the interbody gimbal drive.

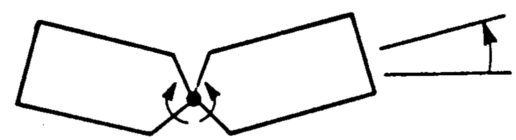
In one study of a system involving the four features noted at the bottom of the chart, the two-body concept was found to be significantly lighter.

TWO-BODY CONTROL

- ALTERNATIVES FOR RAPID REORIENTATION
  - CMG's
  - THRUSTERS
  - TWO-BODY CONTROL







- TWO-BODY CONTROL OFFERS ADVANTAGES FOR SOME APPLICATIONS
  - POINTING WITHIN SEVERAL DEGREES OF NOMINAL POSITION
  - LARGE VEHICLE
  - POINTED BODY LARGE FRACTION OF VEHICLE
  - RAPID REORIENTATION

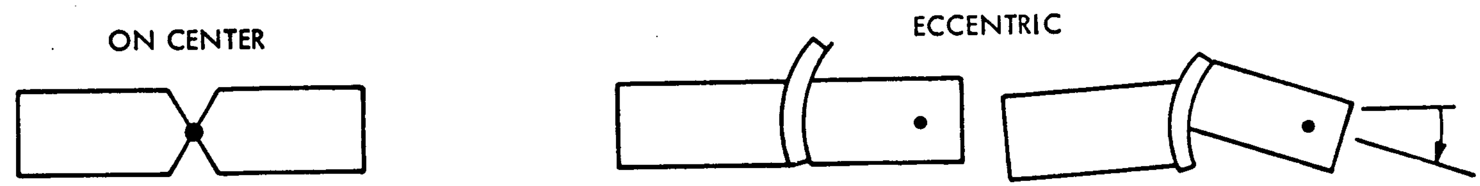
TWO-BODY CONTROL, GIMBAL DEVICES

A variety of gimbal devices are available; and they may be divided into two general categories (Chart 10) - on center (hinge-like) and eccentric. Both translation and rotation take place in eccentric gimbals to provide an apparent pivot point projected forward of the mechanism. Selection of the preferred gimbal device is based on trade studies involving type, pivot location, drive torque, and gimbal relative displacements. One recent study favored an eccentric gimbal.

Applications requiring high-torque reorientation followed by very precise pointing can be accomplished by using the interbody gimbal to produce very large torques together with a smaller CMG for vernier control and precision pointing. (See bottom of Chart 10).

TWO-BODY CONTROL

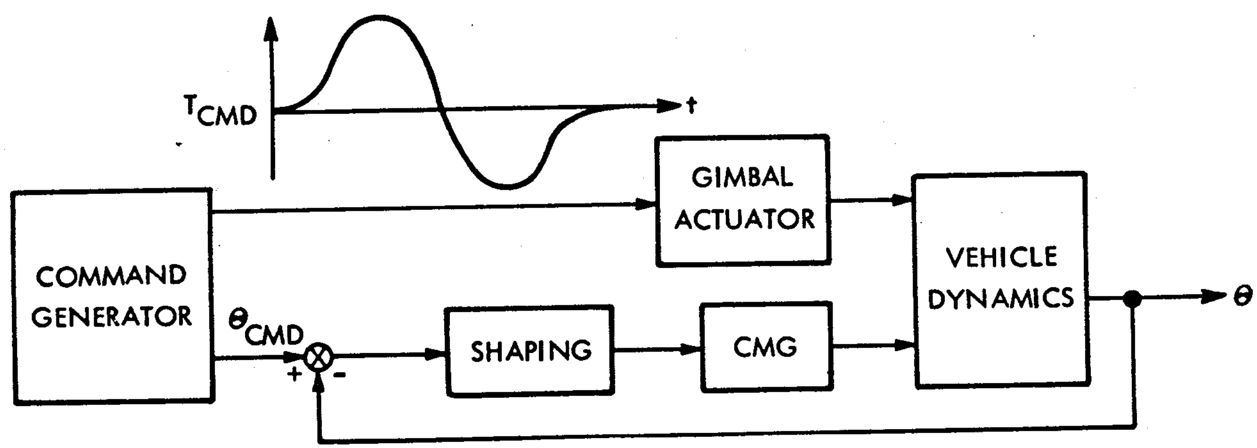
- GIMBAL TYPES



- TRADES

PIVOT LOCATION VERSUS TORQUE, GIMBAL ROTATION, AND GIMBAL TRANSLATION

- LARGE-ANGLE AND PRECISION POINTING

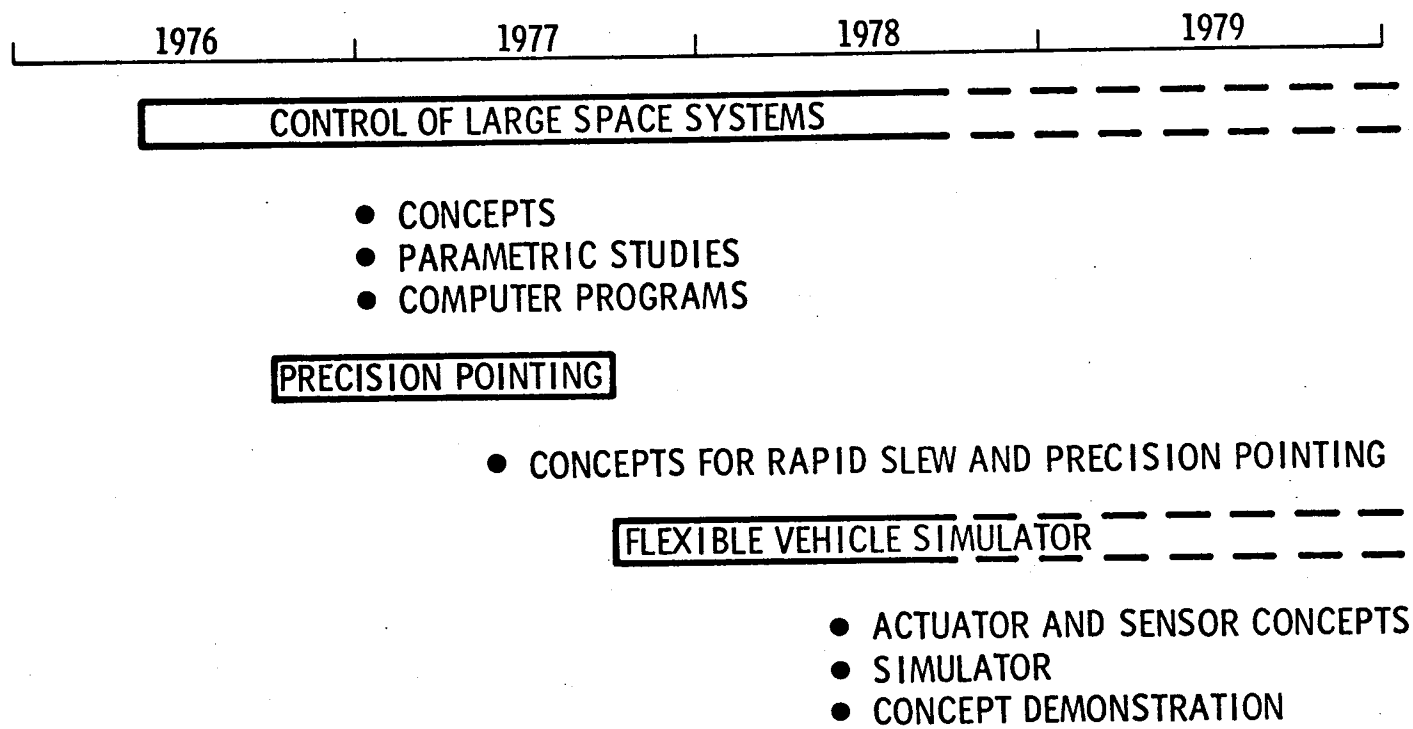


ROCKWELL IR&D PROGRAM

The foregoing discussion highlighted some developments studied in Rockwell's IR&D program. Those concepts were studied in the "Control of Large Space Systems" project, which started in mid-1976, and the "Precision Point" project (Chart 11). So far, the IR&D program has been analytically oriented, but that will soon change with start of new project to construct a flexible vehicle control simulator in the laboratory.

CHART 11

ROCKWELL IR&D PROGRAM FOR CONTROL OF LARGE SPACE SYSTEMS



Large Space Platform Control Avionics Considerations

by Jack G. Fisher

January 19, 1978

Presented at Joint Government/Industry Seminar
on Large Space Systems Technology
at Langley Research Center

Portions of This Data Developed Under Contracts NAS9-15310 and F04701-77-C-0178

GENERAL DYNAMICS
Convair Division

(Figure 1)

General Dynamics/Convair is an active participant in the conceptualization, design and development of several large space programs as shown on the facing page. During the course of this work we have identified a number of areas requiring technology efforts. This presentation identifies some of these areas associated with the avionics-oriented technologies required for design, and operation of many of these large spacecraft.

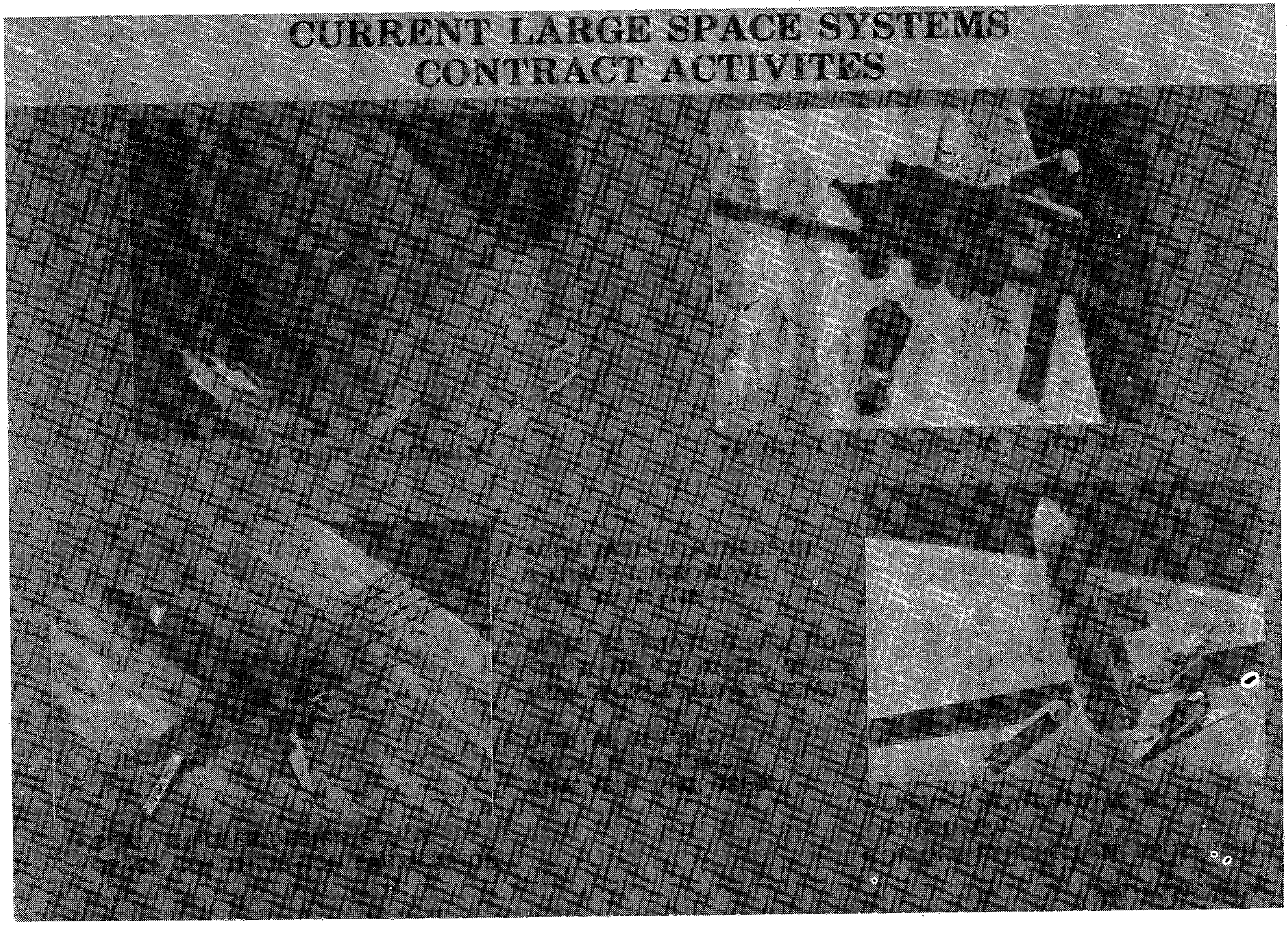


Figure 1

(Figure 2)

A complete list of avionics areas recommended for advanced technology efforts would include:

a) Large structure control and stability analysis and prediction techniques.
b) Rendezvous and docking analysis and simulation tools.
c) Automated positioning and process control method.
d) Analysis of antenna/structural interrelationships, the development of analysis tools, and development of adaptive antenna systems.
e) Electrical power generation/conditioning/and distribution methods adapted for multi-kilowatt systems.
f) Increased emphasis on Orbiter payload support software development.
g) Development of common services accommodations for multiple LSS payload user systems. These would include data management, communications, pointing/stability, power and environmental conditioning functions.

A subset of areas have been selected for more detailed discussion at this time as indicated on the facing page figure. These subjects will be correlated with work being performed under a number of our present LSS Contracts including: OOA, SCAFE, and On-Orbit Propellant Handling.

LSS AVIONICS TECHNOLOGY AREAS

- SIMULATIONS
  - Control & stability
  - Rendezvous guidance
  - Docking
- ROBOTICS
  - Manipulator systems
  - Automated process control
- ANTENNA & STRUCTURAL INTERACTIONS
  - Stiffness vs RF interaction
  - Physical control vs adaptive electronics

Figure 2

(Figure 3)

Two characteristics of large spacecraft systems which differentiate them from conventional satellite systems are:

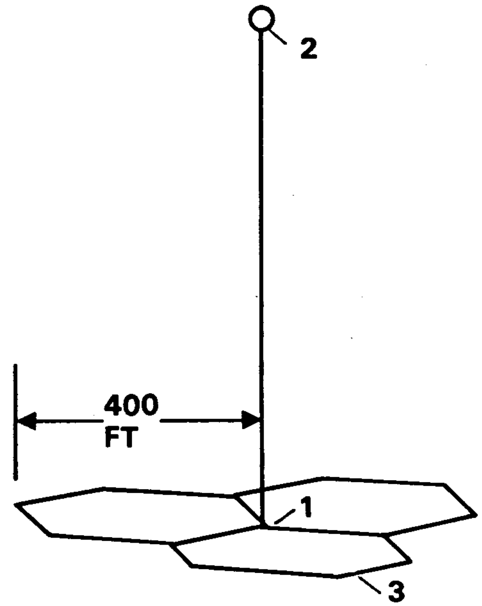
a) They will probably have a distributed avionics system in which a central data management system interfaces with and controls a number of remotely located functions for attitude control or sensing purposes.

b) They represent a class of spacecraft where, because of size or construction, their flexibility and low natural frequencies can interact with the attitude control system.

New techniques for control and stability must therefore be developed along with the analysis and behavior prediction tools to design the spacecraft and properly select its systems components.

The facing page figure shows sample results of one such simulation capability supporting LSS design efforts at General Dynamics/Convair. The spacecraft used in this example was three expandable tetrahedral rings joined together and attached to 1000 ft tower. Each ring is 400 ft by 346.5 ft, and attitude rate data is shown plotted about the roll axis for three widely separated spacecraft locations.

LSS CONTROL SIMULATIONS PREDICT LSS BEHAVIOR
(Dynamic Response to Attitude Control System On-Orbit at GEO)



Attitude & rate sensors
& torquers located at
middle of platform

Control frequency = 0.005 Hz
Structural damping = 0
Initial disturbance = 0.5 deg/sec

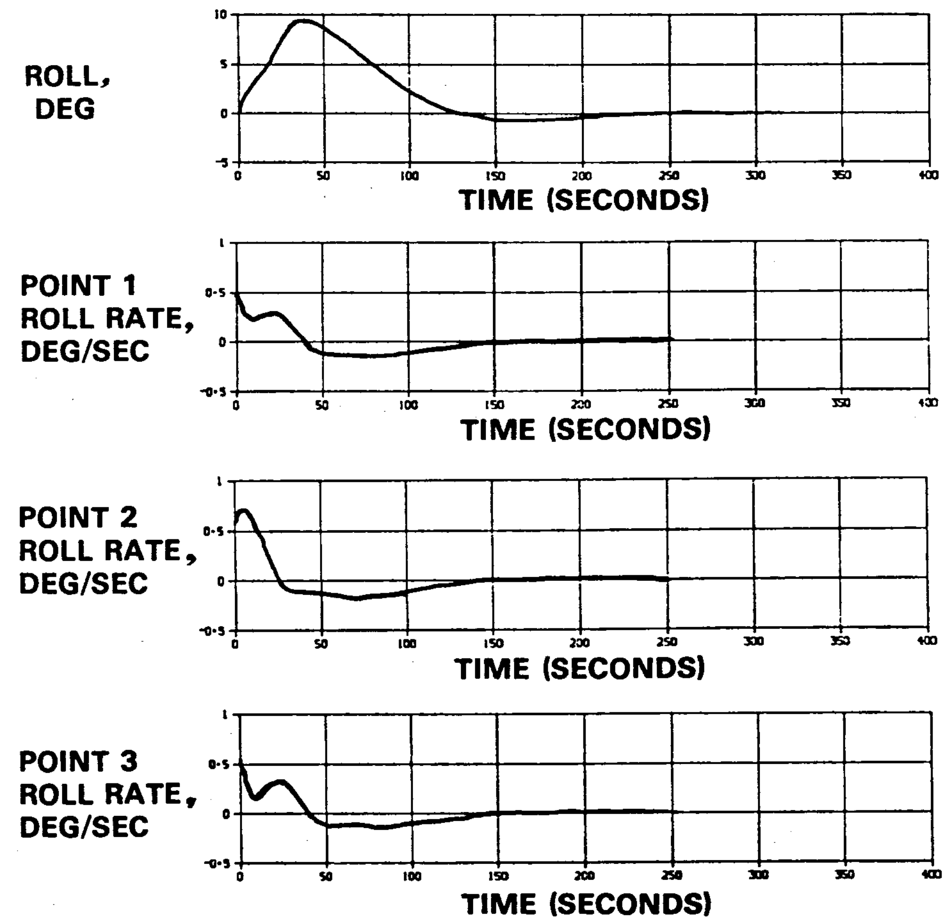
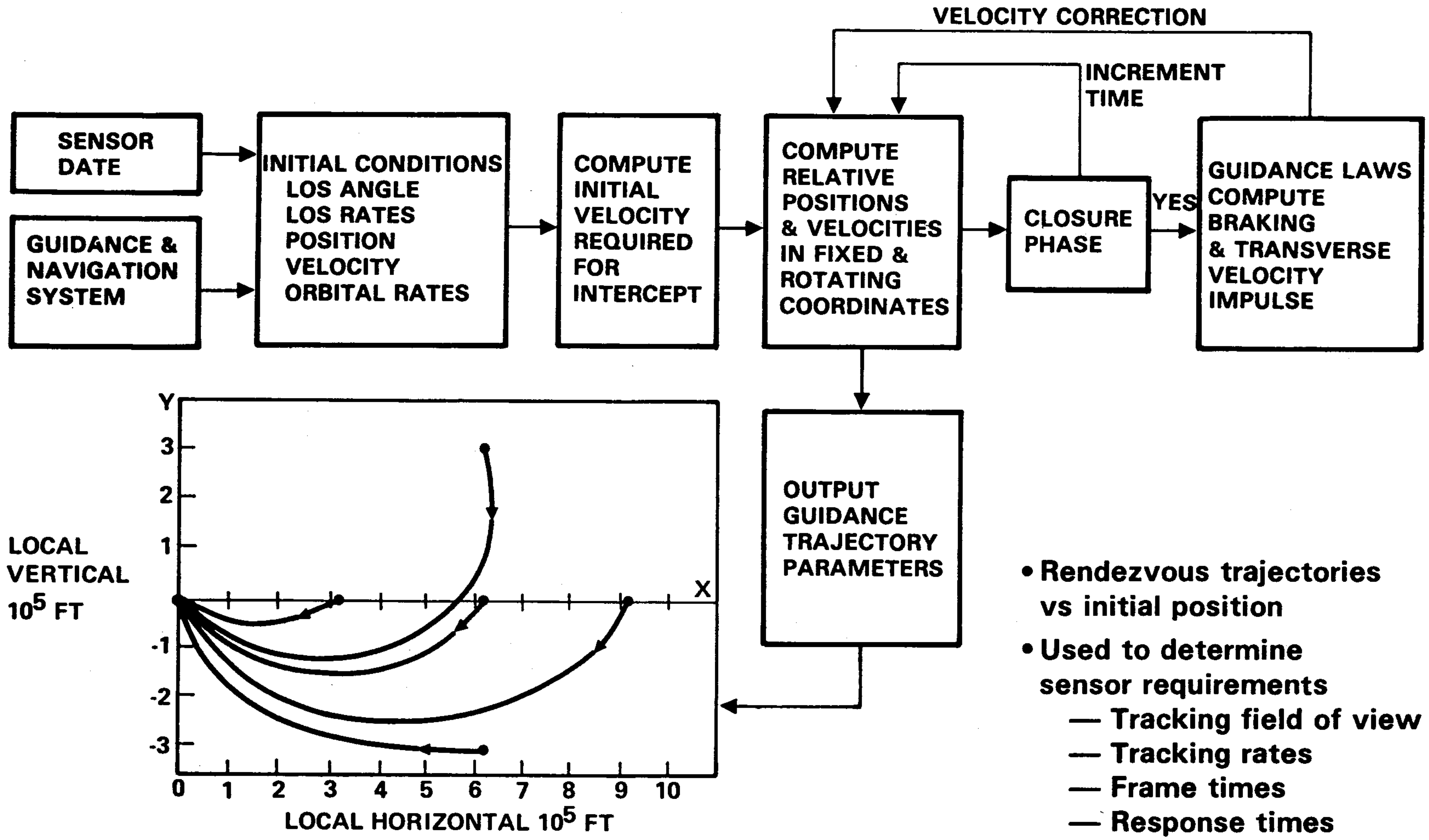


Figure 3

(Figure 4)

Operation or construction of many large space system concepts involves multiple visits by the Orbiter, LSS assembly, or other vehicles. A necessary tool in development of the avionics and propulsion systems to accomplish this function is a simulation of the Rendezvous guidance phase of the Rendezvous and Docking operation. Simulation output of rendezvous trajectories vs. initial position dispersion allow determination of avionic requirements and accuracy characteristics for rendezvous sensors, navigation,and attitude control subsystems.

RENDEZVOUS GUIDANCE SIMULATIONS



- Rendezvous trajectories vs initial position
- Used to determine sensor requirements
  - — Tracking field of view
  - — Tracking rates
  - — Frame times
  - — Response times

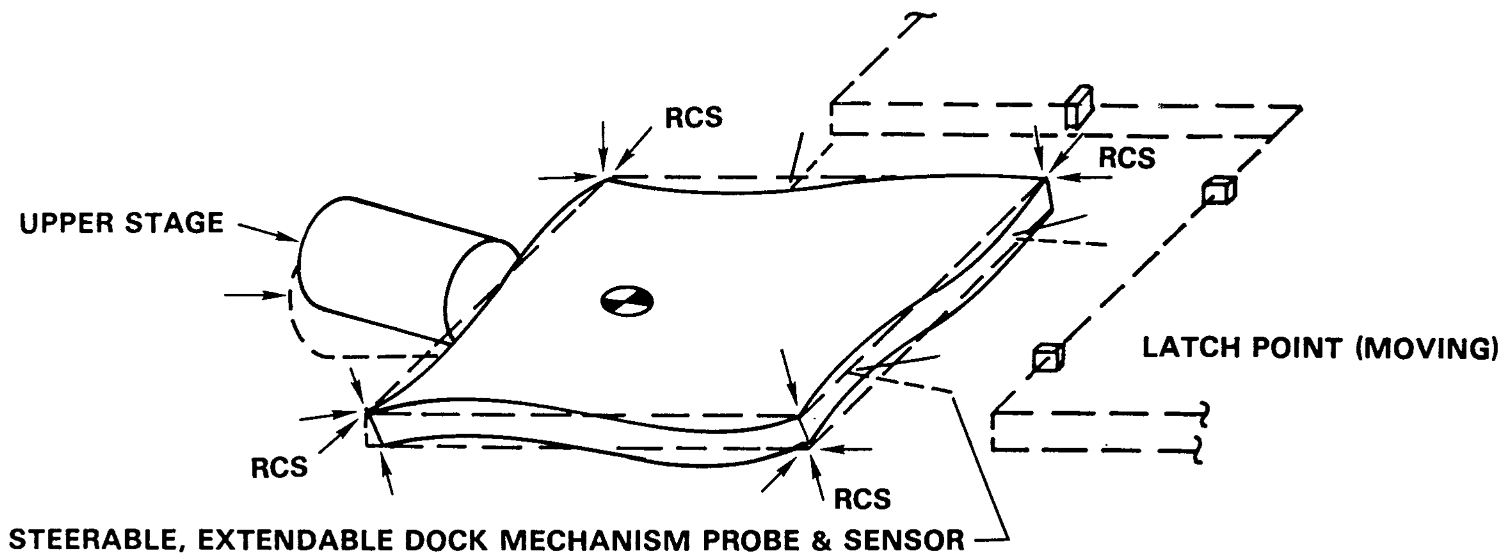
Figure 4

(Figure 5)

The second phase of the multiple visit operation associated with many large spacecraft systems under consideration involves terminal phase docking between one LSS and another or between a LSS and a service vehicle such as the Orbiter or orbit transfer vehicle. Unlike docking operations in previous space programs, LSS docking will likely be characterized by a) remote control, b) soft docking techniques and c) difficulties caused by flexible structure/attitude control interactions.

The figure illustrates this problem peculiar to remote assembly of two large highly flexible spacecraft modules. In this example, the chase/upper stage module is active while the primary or mother module is cooperative but essentially inactive. Simulations of these operations will allow determination of LSS subsystem requirements for docking operations including attitude control and stability functions, docking sensors, operational techniques and timing, and docking mechanisms. The effects of communication delays for control/monitoring purposes and vehicular dynamic interactions during latch-up may also be determined.

LARGE SPACECRAFT DOCKING



- Associated with many LSS operations
  - — Orbiter-to-LSS docking
  - — In-space assembly or servicing
- Characterized by
  - — Remote control
  - — Soft docking
  - — Flexible structures

Figure 5

(Figure 6)

The figure on the opposite page shows the basic elements of a two body docking simulation involving large spacecraft systems. It takes into account the physical model (rigid and flexible body characteristics) of both chase and mothercraft, initial position and velocity conditions, docking sensor and autopilot models and the effects of environmental influences. The latter include forces due to solar wind, solar heating, gravity gradient, and magnetic torques. For the simulation shown, the chase craft is being maneuvered into a position, relative to the mother, suitable for docking. This chase attitude and translation position occur when all docking mechanisms are at their neutral position and ready to latch. This situation must be maintained, within control system limits, by chase RCS control while the docking mechanism latch point is steering into position and latched and despite RCS limit cycling, chase flexure and mothercraft motion.

Just after latch the simulation dynamic situation significantly changes. Contact forces are applied between modules and can cause control problems. Presently we envision turn-off of the chase autopilot for neutralization of docking mechanism forces with the mother still under its autopilot control as an appropriate control system concept for post-latch.

DOCKING SIMULATION BLOCK DIAGRAM

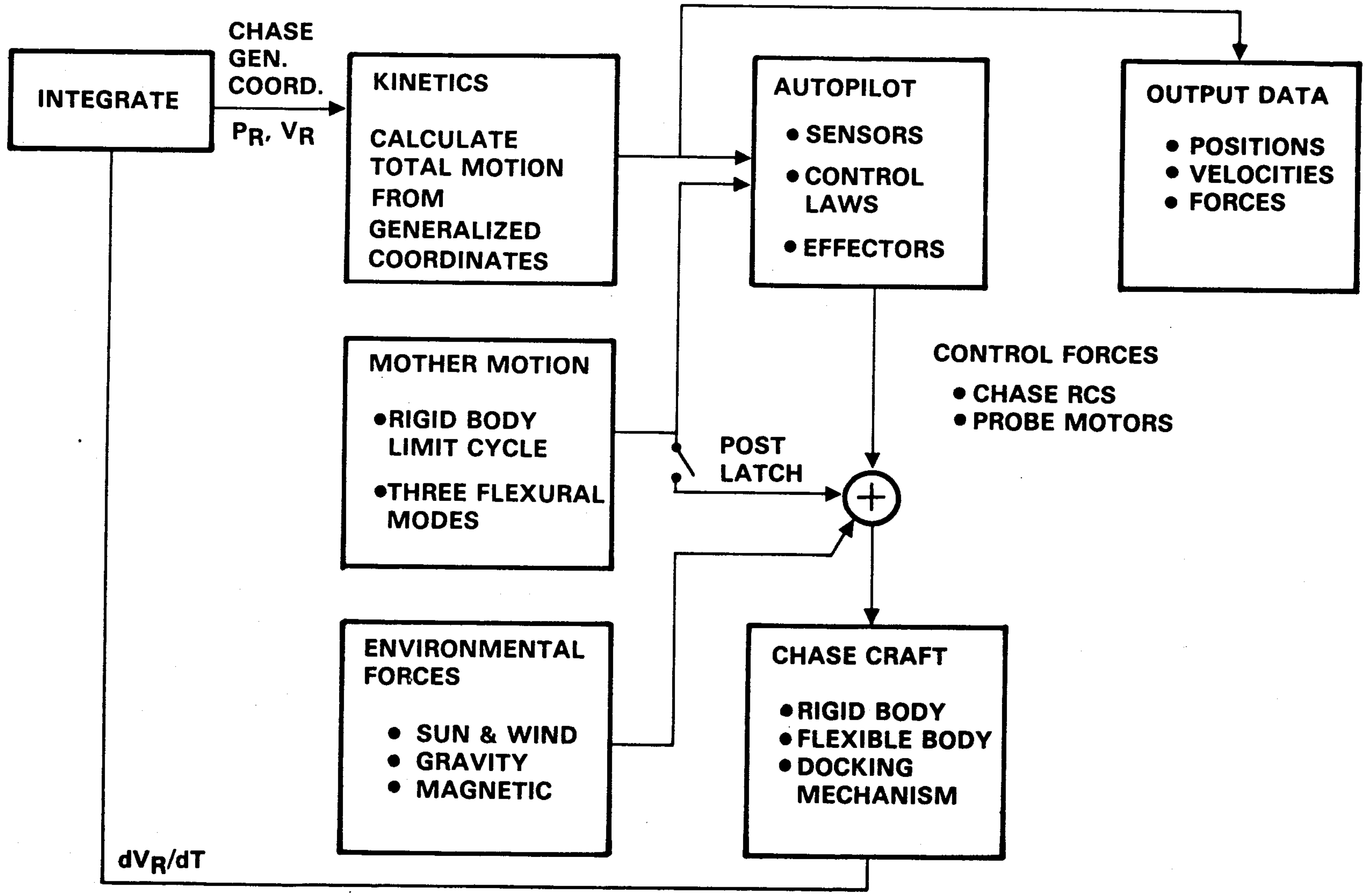


Figure 6

(Figure 7)

The baseline Space Construction Automated Fabrication Experiment (SCAFE) system is shown in the illustration. It represents a major step in utilization of space as a working environment, and demonstrates one technique for implementing Large Space Structures. In this technique automated fabrication/assembly systems and prepackaged raw materials are delivered by Shuttle to the construction circular orbit at 300 nautical miles (556 km).

Upon system deployment from the stowed position, a beam builder, moving to successive positions along a Shuttle-attached assembly jig, automatically fabricates four triangular beams, each 200 meters long. Retention and manipulation of the completed beams is provided by the assembly jig.

The beam builder then moves to the position shown and fabricates the first of nine shorter, but otherwise identical, crossbeams. After crossbeam attachment, the partially completed assembly is automatically transported across the face of the assembly jig to the next crossbeam location, where another crossbeam is fabricated and installed. This process repeats until the "ladder" platform assembly is complete. During this process an opportunity to develop/evaluate EVA is provided by the difficult-to-automate task of sensor/equipment attachment, as shown.

SPACE CONSTRUCTION AUTOMATED FABRICATION EXPERIMENT DEFINITION STUDY (SCAFEDS)

GENERAL DYNAMICS
Convair Division

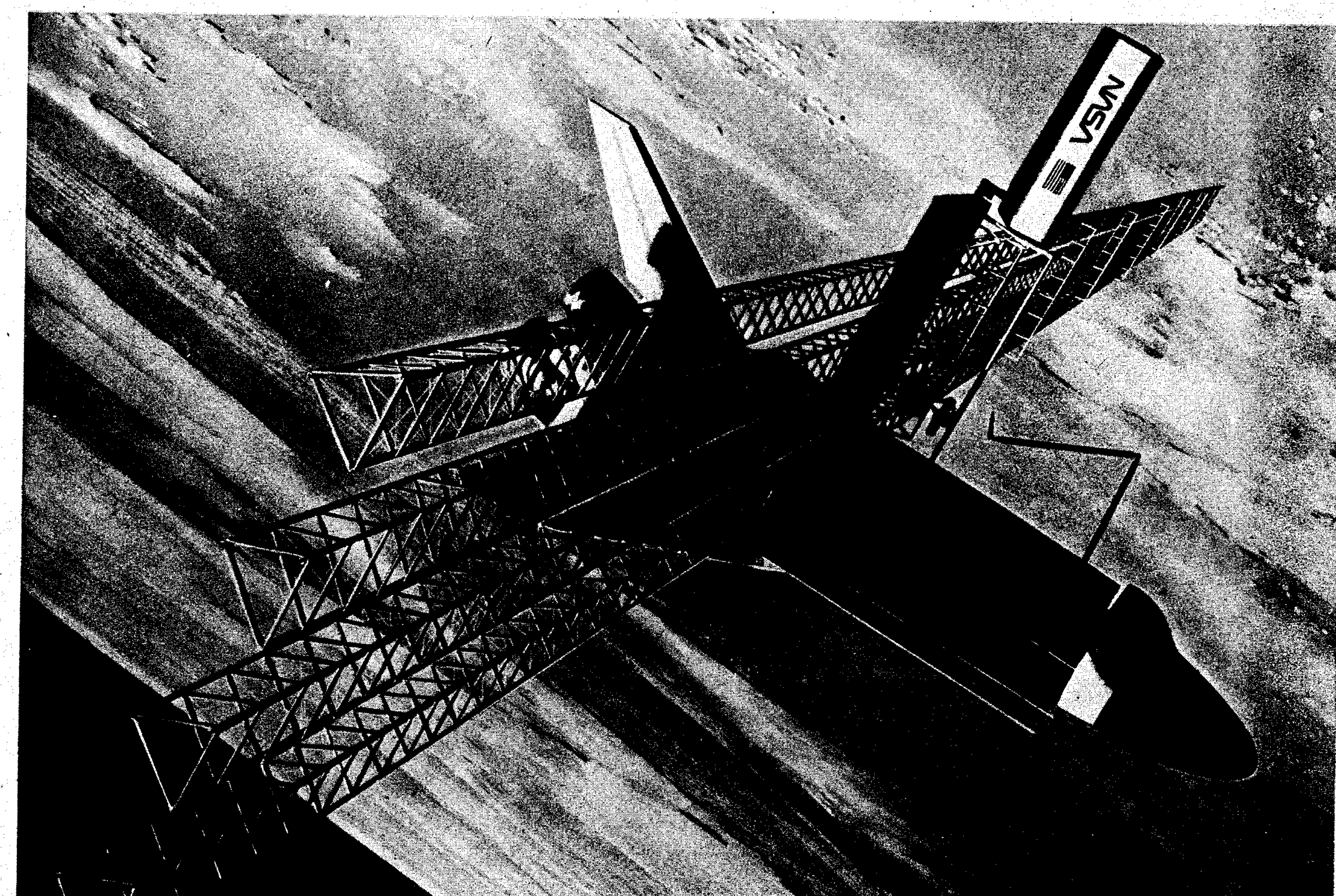


Figure 7

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(Figure 8)

This conceptual drawing serves to illustrate the arrangement and function of the selected beam builder configuration. Automated beam builder machine functions under the executive control of the Orbiter crew and major configuration features are summarized as follows:

Machine Operations

- Storage - Flat continuous strips wound in rolls, cord wound on spools, cross members preformed and precut and stored in clip feed mechanism.
- Heating - Electrical resistance wire reflective strip heaters.
- Forming - Rolltrusion.
- Cooling - Fluid cooled platens.
- Drive - Friction roller drive.
- Cross member Positioner - swing arm, single drive.
- Cord Positioning and Pretensioning - Reciprocating cord plyers on reversing screws with constant force tensioning mechanisms.
- Joining - Ultrasonic spot weld head.
- Cutoff - Shears.

SCAFEDS BEAM BUILDER CONCEPT

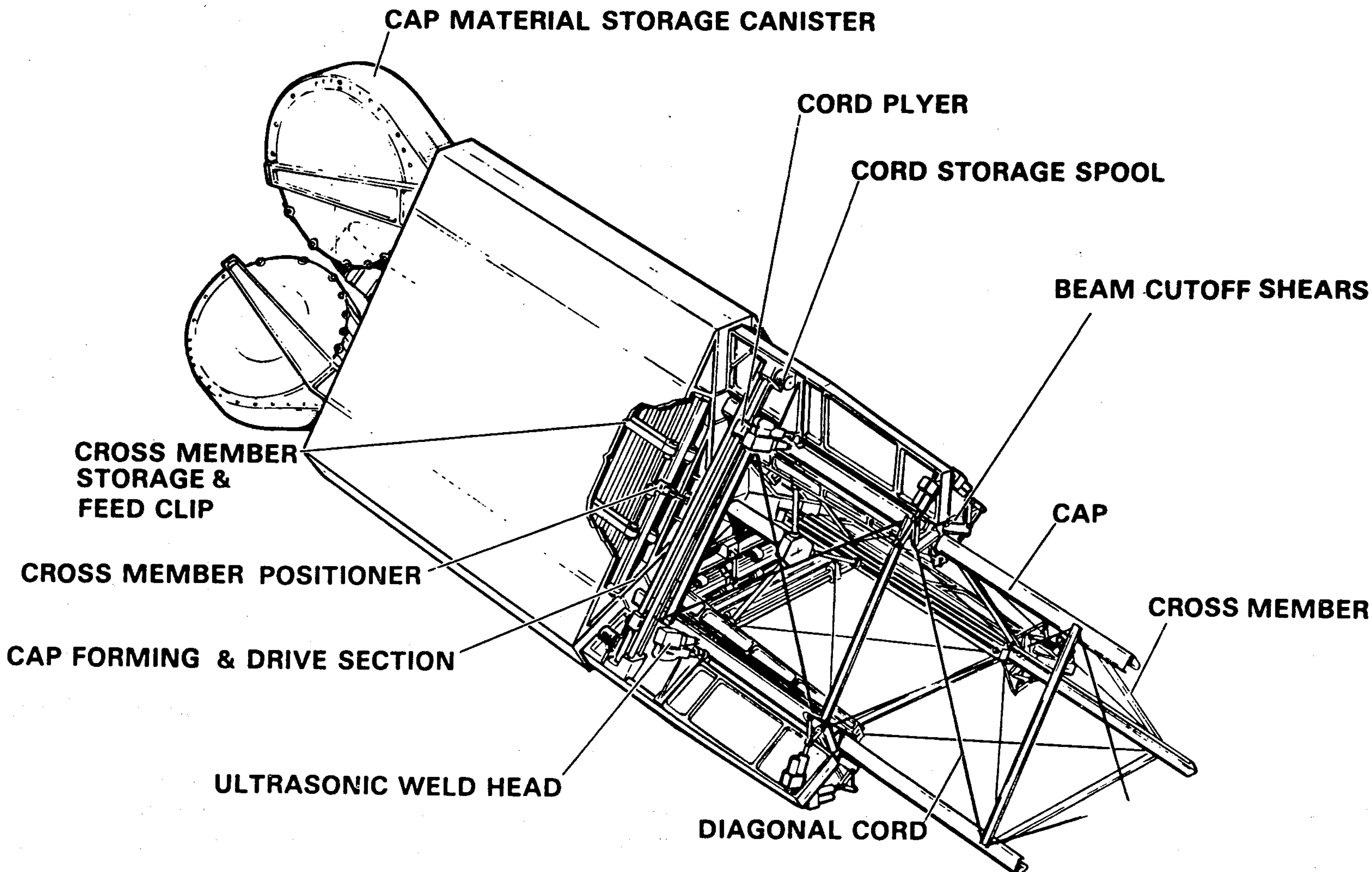


Figure 8

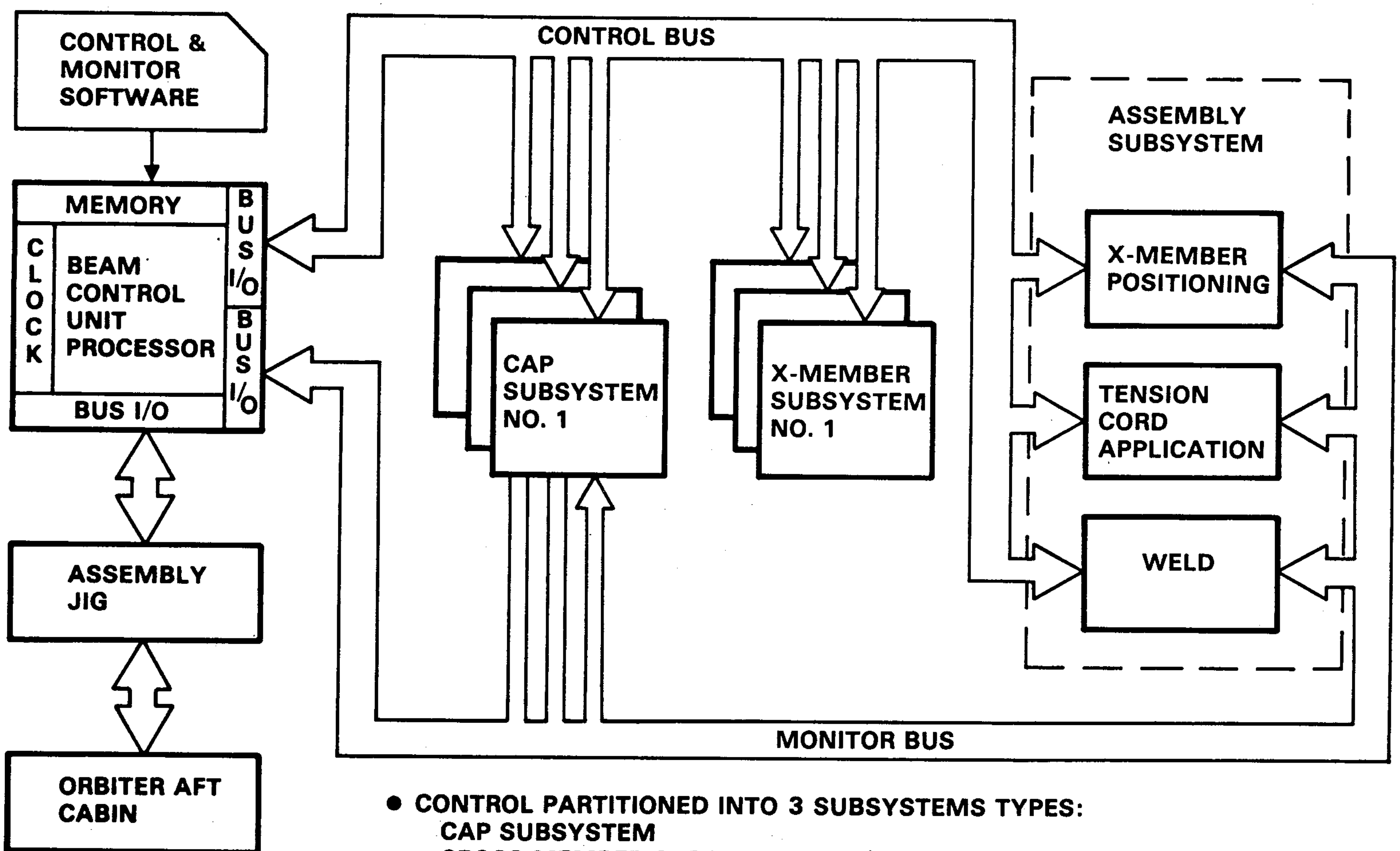
(Figure 9)

The beam builder control and monitor system is housed within the beam builder machine and provides the real-time process control operations to fabricate beam elements to the correct length and straightness.

This system is made up four major subsystem functions as shown in the figure. These are the: (1) beam control unit, (2) cap fabrication subsystem, (3) cross member positioning subsystem, and (4) beam fabrication subsystems. The Beam Control Unit (BCU) performs overall control and monitoring of beam builder operations and contains a microprocessor with interval timer, approximately 4K of memory and input-output interfaces to the assembly jig and the three fabrication control systems. It receives executive control from the Orbiter crew via the Orbiter MDM/data bus interface located on the assembly jig and provides status data back to the crew.

The cap subsystems contain those motors/actuator controls and sensors necessary to control fabrication of the three cap member sections. The cross member subsystems contain the motor/actuator control and sensors to remove the preformed cross members from the storage clips, and position them for final assembly, apply the tension cords and finally to weld the beam components together into one integral bay length unit.

BASELINE BEAM BUILDER CONTROL SYSTEM



- CONTROL PARTITIONED INTO 3 SUBSYSTEMS TYPES:
  CAP SUBSYSTEM
  CROSS MEMBER SUBSYSTEM
  ASSEMBLY SUBSYSTEM
- BCU FUNCTIONS SIZED FOR 8-BIT MICROPROCESSOR

Figure 9

(Figure 10)

Typical beam builder automatic process control operations are shown in the illustration for the cap control subsystem. In this subsystem, control and monitor functions are provided to process flat graphite thermoplastic material from the storage reel into formed structural members of the desired length. Upon entering this subsystem the material is heated to 425°F in the heating section and maintained at that temperature in the forming sections. Heating is accomplished using four sets of helically wound elements connected to individual control unit under BCU control. IR-type temperature sensors will be monitored for this purpose.

From the heaters the material passes into the cooling section where platens will be commanded closed for 38 seconds during the assembly portion of the cycle.

Cap length control is the basis of beam alignment control for the baseline configuration. As such, each cap subsection will contain dual redundant variable speed beam drive motor units. In operation a travel sensor system with a resolution of 0.25 mm will provide cap length data for each cap to the BCU processor. The final step of the cap fabrication system occurs when the desired total beam section length has been produced. On BCU command the three cap subsystem shear mechanisms will be activated.

Areas recommended for continued technology development to support this and similar efforts are as follows:

- Specific Recommendations
  - Heater elements and temperature control
  - Beam alignment control subsystem
  - Ultrasonic weld process control
  - Platform instrumentation/experiment accommodations

- General Recommendations
  - Design/selection of space qualified limit sensors
  - Design/selection of mechanism drive motors
  - Detailed automation timing/control software development

CAP MEMBER
SUB-SYSTEM CONTROL DIAGRAM

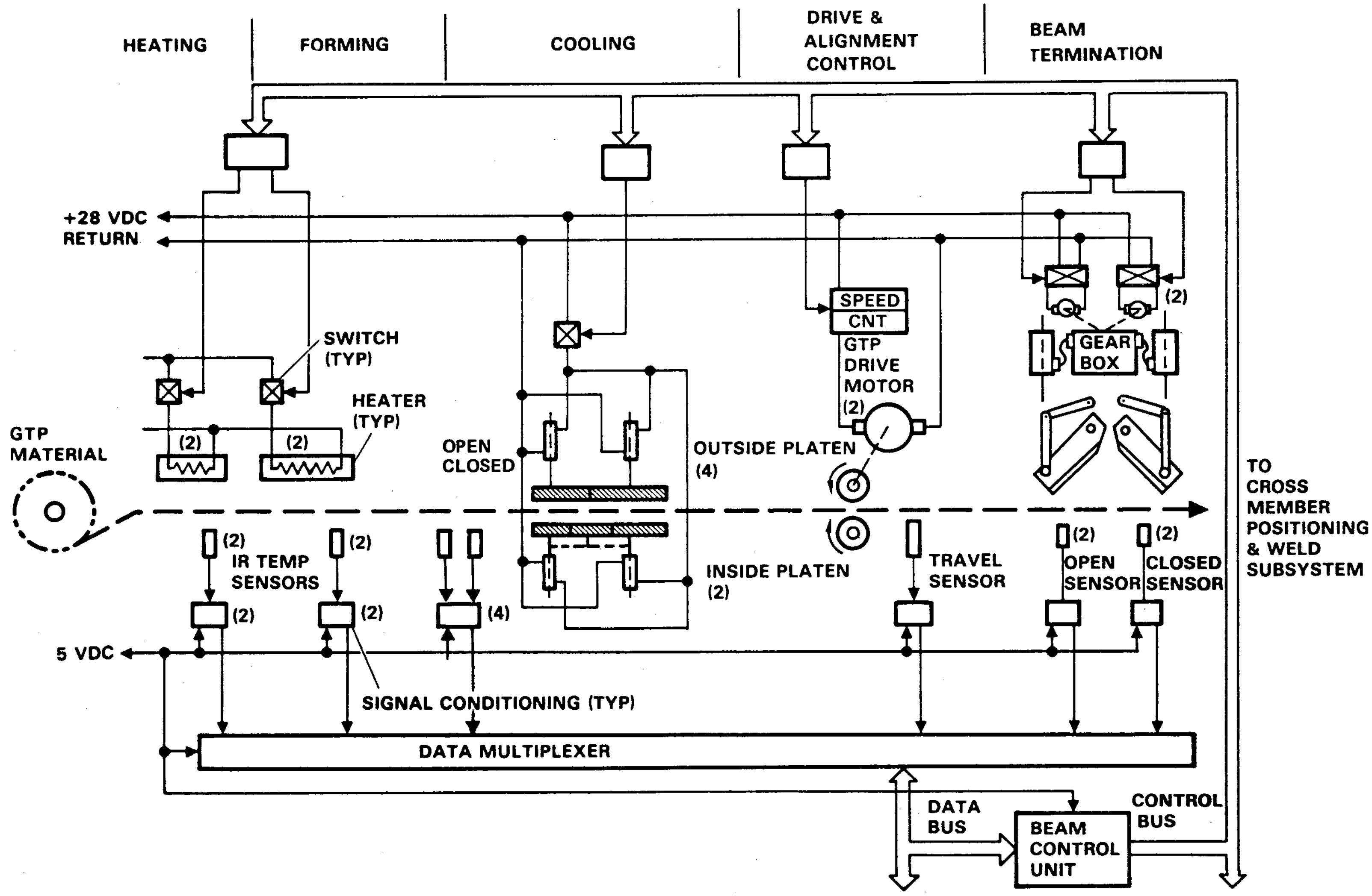


Figure 10

(Figure 11)

During the early 1980's space activities will shift from expendable launch vehicles to the Space Shuttle. This is expected to be followed by the introduction of larger reusable vehicles (Orbital Transfer Vehicle) and satellites which will require replenishment of expendables.

Obtaining the maximum benefit from this new generation of spacecraft will require propellant supply services that cannot easily be provided by standard Shuttle payload accommodations. General Dynamics, in 1976 began conceptual work for alternate propellant delivery techniques which capitalize on Shuttle's contingency payload capability.

With this approach, excess Shuttle payload capability is consigned to water stored in tanks beneath the Orbiter's payload bay. Water is a suitable Shuttle contingency payload since it is relatively dense, has very simple container requirements, and causes no Shuttle safety concern. This water is delivered to an orbiting processor as an ancillary payload during deployment of dedicated mission payloads. The propellant processor uses solar energy to reduce the water to gaseous hydrogen and oxygen, liquefy the propellants, and reliquefy boiloff from the storage tanks. Using this method, the propellants can be accumulated in low earth orbit for use by other vehicle or satellite systems.

ON-ORBIT PROPELLANT PROCESSING

GENERAL DYNAMICS
Convair Division

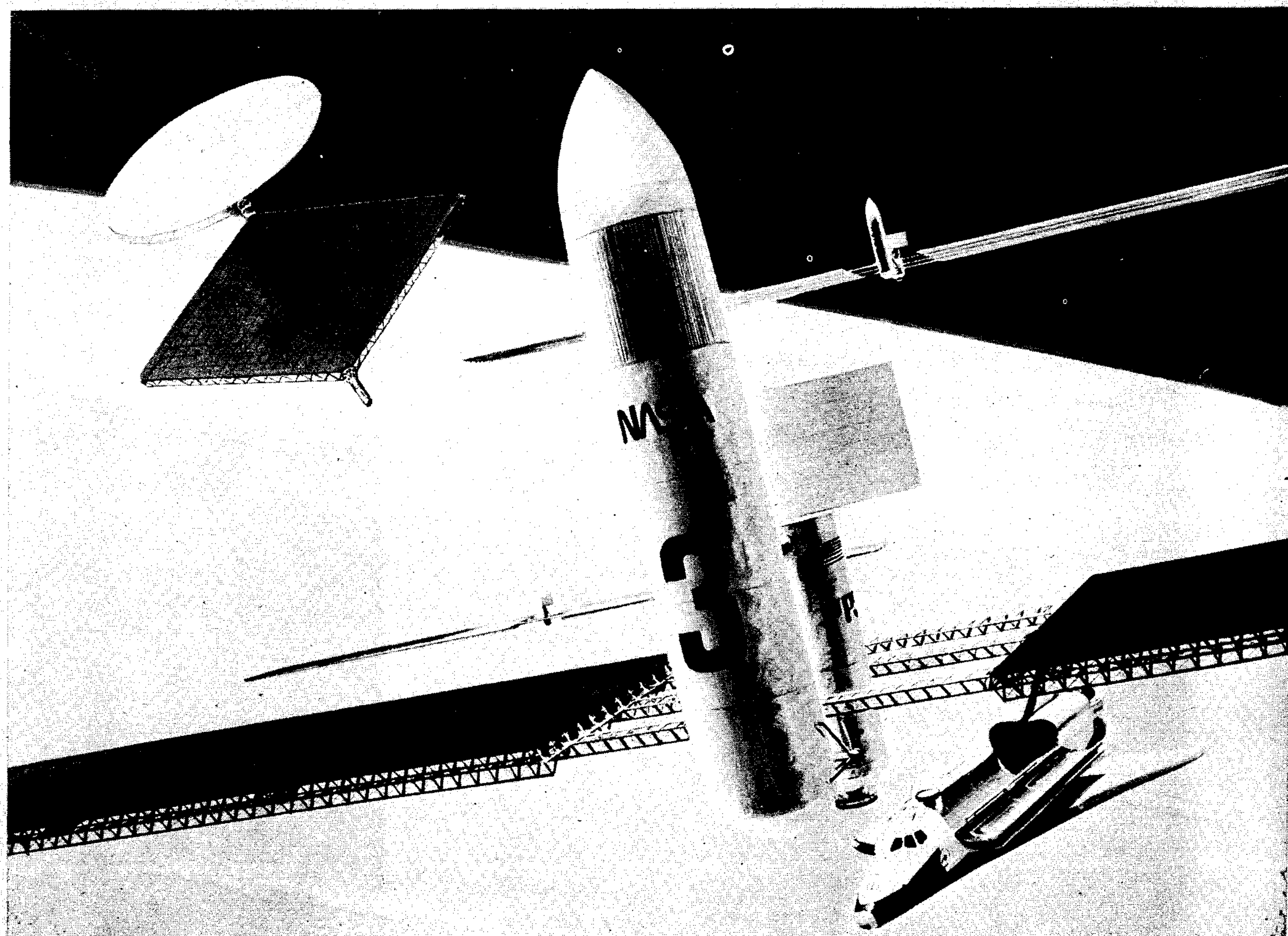


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Figure 11

(Figure 12)

The propellant processor represents a large spacecraft system which will pace the technology development required for the majority of large space structure applications. Avionics subsystems utilized include: (1) power monitoring and control, (2) propellant processor monitoring and control, (3) processor stationkeeping, (4) processor attitude and solar panel pointing control, (5) rendezvous and docking, (6) communications and data management, and (7) safety. These functions are shown in the block diagram of the facing page figure.

Propellant processor control functions encompass water storage/supply control, electrolysis control, liquefaction control, and propellant storage control. These functions will be dependent upon requirements resulting from command and monitoring data rates, response times, and software for data processing, storage, and control.

Propellant processor stationkeeping, stability, pointing and attitude control requirements are important for determining operational phases and control functions. These include solar array sun tracking, radiator panel pointing (perpendicular to solar vector), and rendezvous and docking control/stability. Attitude control and data management systems and associated sensors for performing these functions are candidate areas for future technology development.

PROCESSOR AVIONICS SYSTEM ELEMENTS

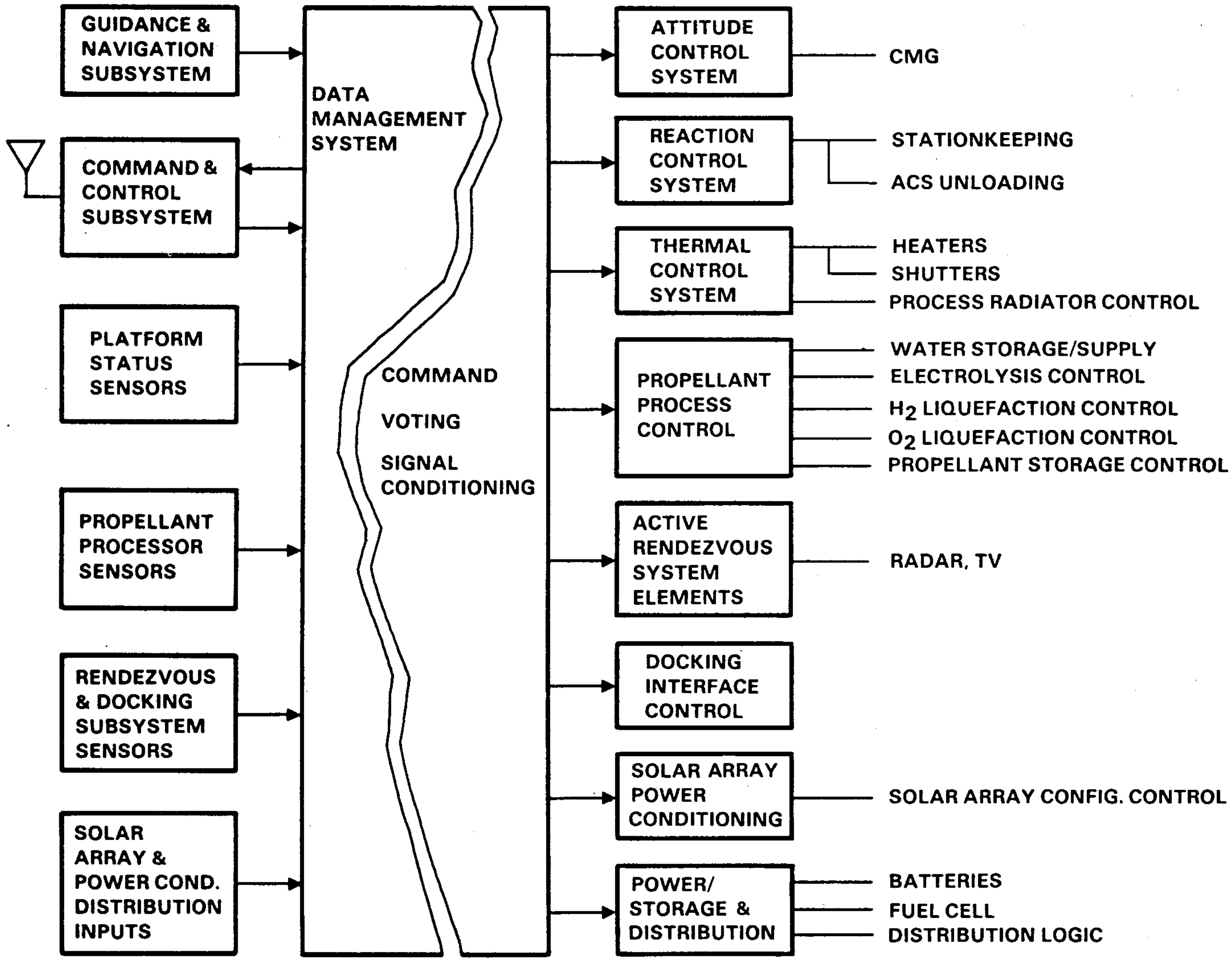

Figure 12

(Figure 13)

One of the more exciting applications of Large Space Structures technology is the development of very large space-based antenna systems. The illustration shows one large antenna concept based on an active lens phased array technique. New technology effort based on these LSS applications include:

a. Development of antenna designs compatible with LSS antenna deployment or assembly techniques.

b. Development of methods for distribution of power and control networks which have low impact on assembly and operation.

c. Development of antenna elements and systems which track and adopt to structural warpage and dynamic affects.

d. Analysis and simulations associated with the interactions between structural characteristics, the antenna element implementation and resulting RF/antenna operating performance. These include effects due to structure depth and blockage, surface control or element misalignment, and grating lobes vs antenna gap or irregularity.

LARGE ANTENNA CONCEPT BASED ON LSS

- TECHNOLOGY EFFORTS REQUIRED IN
  - — DEPLOYMENT CONTROL
  - — POWER DISTRIBUTION
  - — ADAPTIVE RF ELEMENTS
  - — STRUCTURAL/ANTENNA INTERRELATIONSHIPS

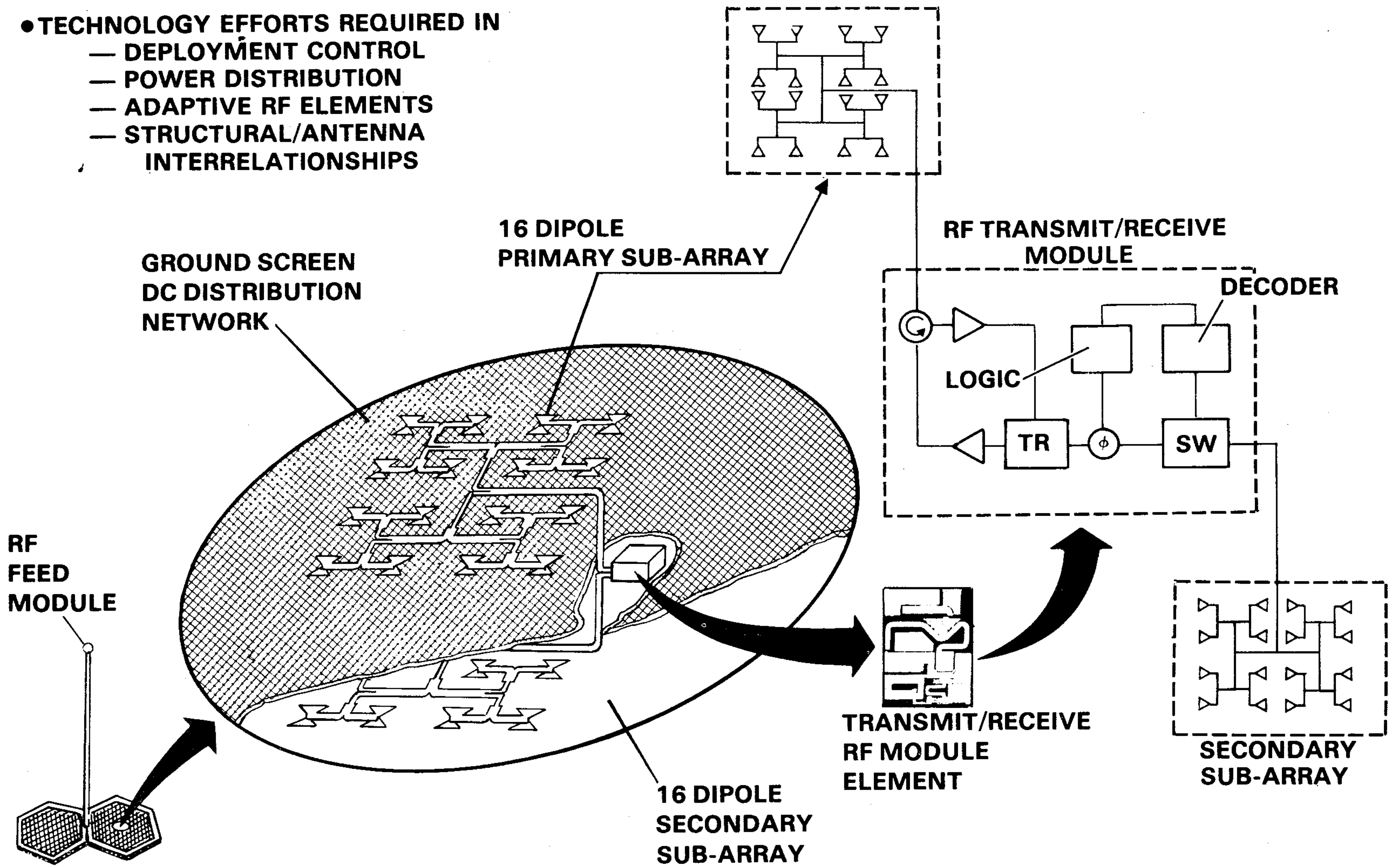

Figure 13

(Figure 14)

Our efforts in the concept development and design of future large spacecraft systems has resulted in the identification of new technologies peculiar to development or functional operation of these systems. A number of these have been discussed in the preceding material and are summarized in the table.

SUMMARY
CONTINUED TECHNOLOGY EMPHASIS

- Large structure system behavior analysis prediction tools
- Automated process control
  - — Sensors
  - — Actuators & mechanisms
  - — Welding & joining techniques
- Antenna systems development
  - — Surface control
  - — Analysis
  - — Deployment techniques

Figure 14

Maneuvering and Pointing Flexible Vehicles

Douglas C. Fosth
Boeing Aerospace Company
Seattle, Washington

Presented At
Large Space Systems Technology Seminar
Langley Research Center

January 17-19, 1978

THE PROBLEM (Figure 1)

WITH THE DEVELOPMENT OF TECHNIQUES TO ASSEMBLE LARGE STRUCTURES IN ORBIT, NEW CONTROL SYSTEM PROBLEMS EVOLVE. THESE LARGE STRUCTURES ARE TYPICALLY CHARACTERIZED BY LOWER STRUCTURAL FREQUENCIES, BUT NO COMPROMISES ARE MADE REGARDING MANEUVER AND STRUCTURAL SETTLING TIMES. TECHNIQUES MUST BE DEVELOPED WHICH WILL ALLOW THESE LARGE STRUCTURES TO BE MANEUVERED AND POINTED QUICKLY WITH MINIMUM SETTLING TIMES.

The Problem

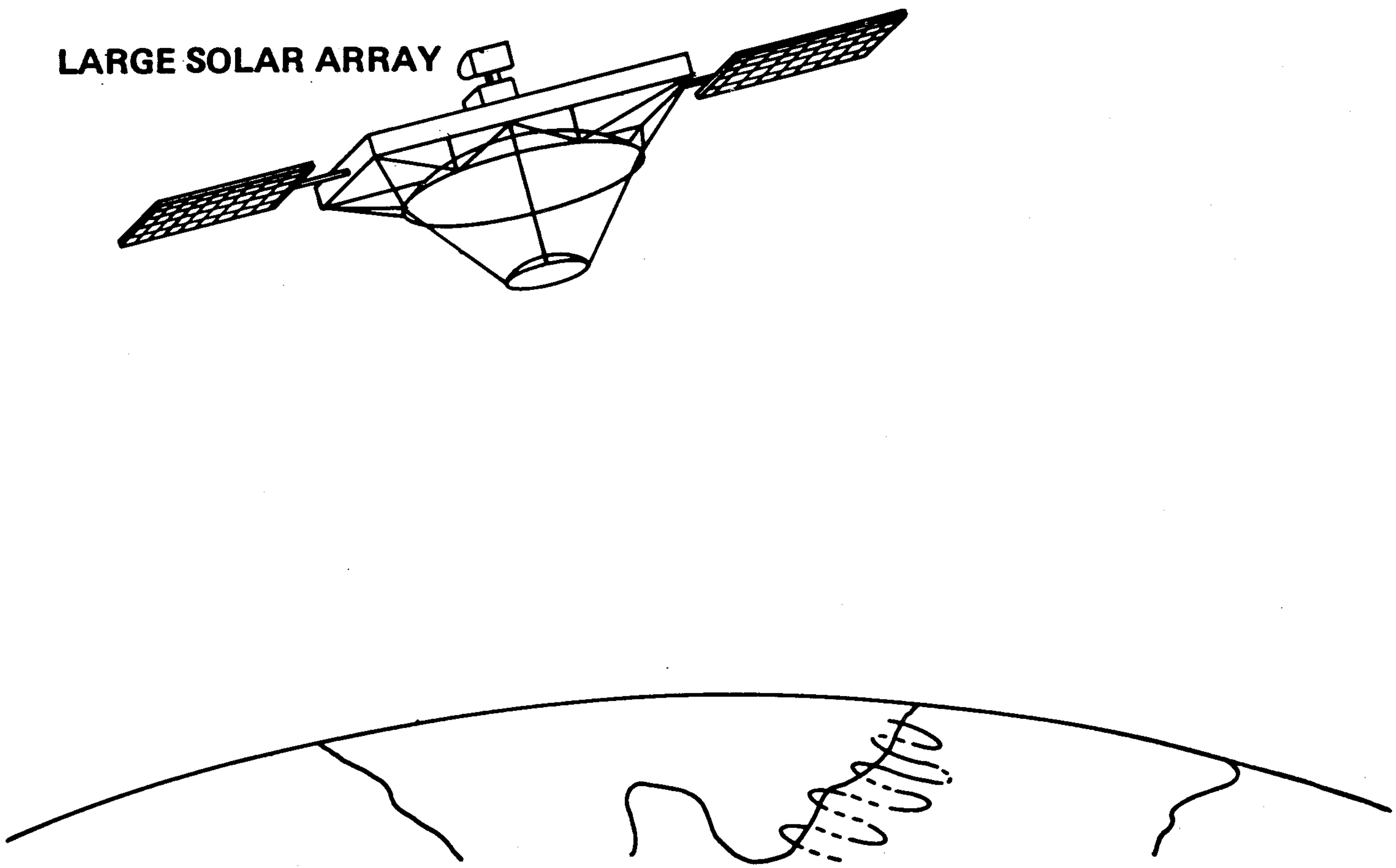

Figure 1

TEXT

APPROACH (Figure 2)

BOEING USES A THREE-PRONGED APPROACH TO THIS PROBLEM. IN THE PRE-CONTRACT AWARD TIME PERIOD, ANALYSIS AND SCALE MODEL TESTS ARE USED TO DEVELOP AND VERIFY CONTROL SYSTEM DESIGN CONCEPTS; AFTER CONTRACT AWARD FULL-SCALE TESTS ARE USED IN THIS ACTIVITY WITH THE PURPOSE OF PROVIDING LIMITED PROOF OF PERFORMANCE AS WELL AS DETAILED DATA ON SYSTEM INTEGRATION.

Approach

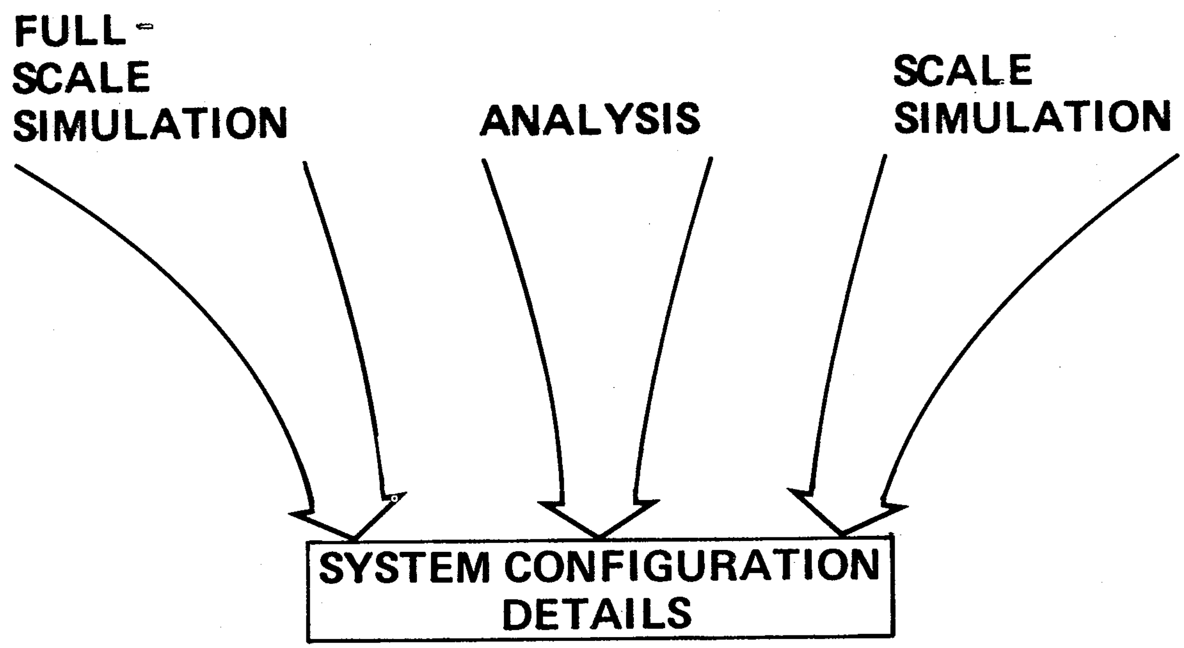

Figure 2

TEXT

MANEUVER APPROACH (Figure 3)

INITIALLY, THE ACCEPTED METHOD OF MANEUVERING VEHICLES WAS TO SLEW EACH VEHICLE AXIS AT THE RATE LIMIT CAPABILITY OF THE SYSTEM AND ACCEPT THE SETTLING TIME PENALTY. WITH THE ADVENT OF NEW COMPUTER TECHNOLOGY, A DIFFERENT APPROACH CAN BE USED WHICH IS BASED UPON GENERATING A VIRTUAL TARGET FOR THE SPACECRAFT TO EXECUTE DURING EACH SLEW MANEUVER. THIS METHOD PRODUCES SMALL INCREMENTAL ANGLE AND RATE CHANGES IN A PRE-PROGRAMMED MANNER WHICH HAS THE EFFECT OF MINIMIZING THE SIZE OR MAGNITUDE OF THE CONTROL ERROR SEEN BY THE VEHICLE.

Maneuver Approach

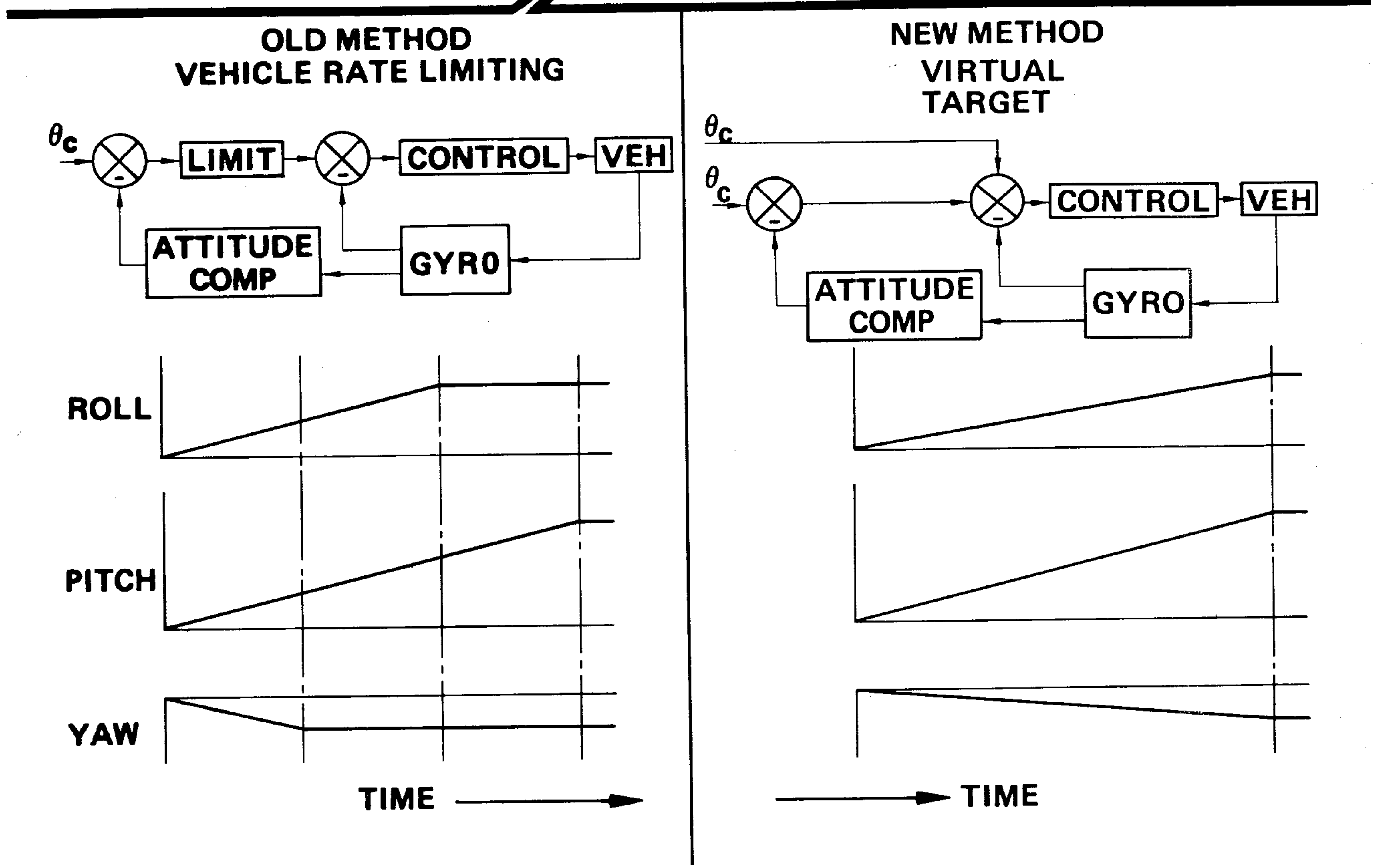

Figure 3

.996 TEXT

VIRTUAL TARGET APPROACH (Figure 4)

THE USE OF THE COMPUTER TO GENERATE THE VIRTUAL TARGET ALLOWS FLEXIBILITY IN SELECTING THE METHOD USED TO GENERATE THE DESIRED TRAJECTORY. IT HAS BEEN DETERMINED, BASED UPON BOTH ANALYSIS AND TEST, THAT IMPOSING A JERK LIMIT (RATE OF CHANGE OF ACCELERATION) AND AN ACCELERATION LIMIT UPON THE TRAJECTORY WILL NOT CAUSE EXCESSIVE STRUCTURAL EXCITATION. THIS IS FEASIBLE BECAUSE THE IMPOSED LIMITS ARE JUST NUMBERS FIXED IN THE COMPUTER PROGRAM WHICH GENERATES THE VIRTUAL TARGET.

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Virtual Target Approach

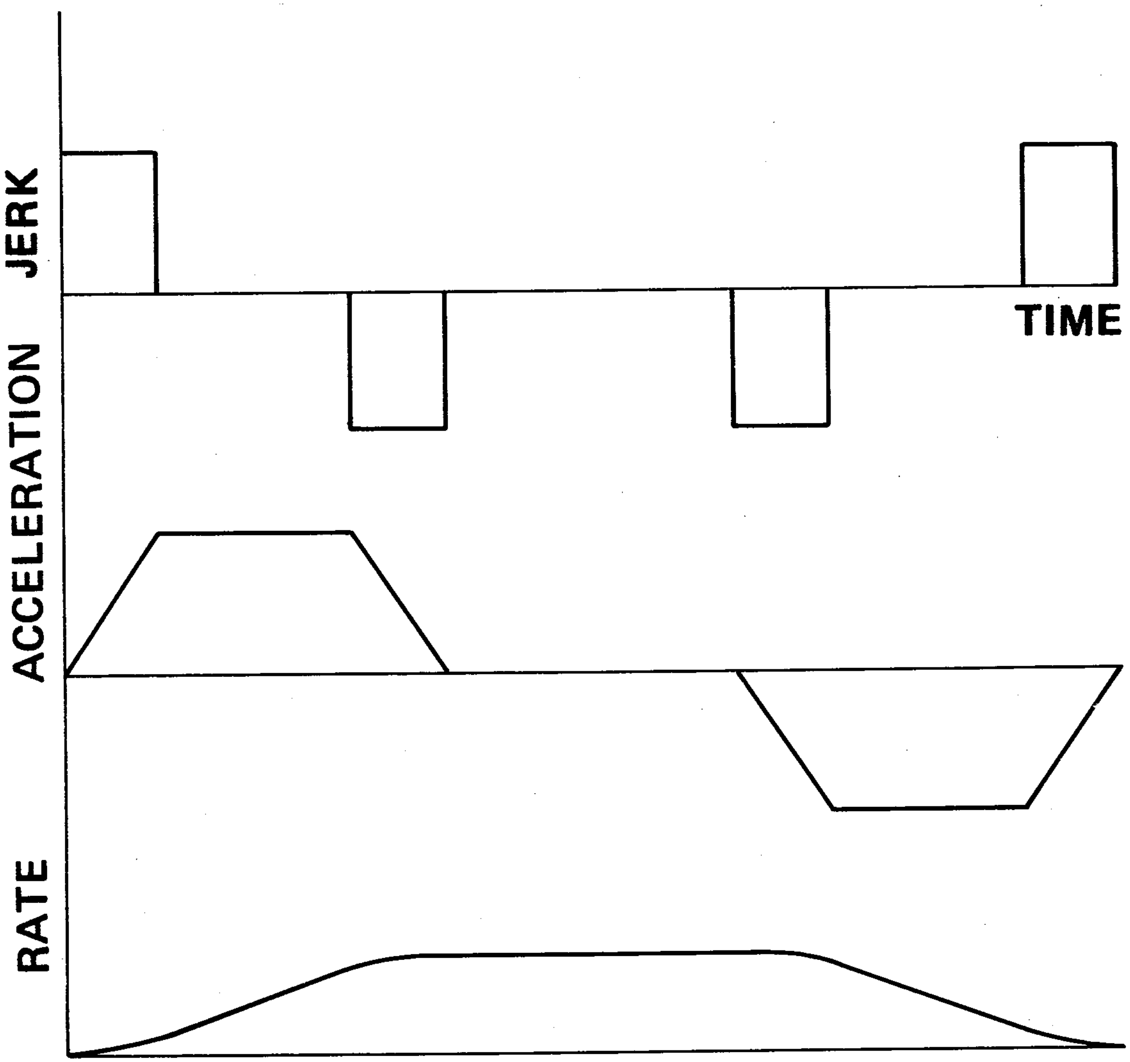

Figure 4

LOW FREQUENCY STRUCTURE RESULTS (Figure 5)

FOR STRUCTURAL FREQUENCIES GREATER THAN FIVE HZ, IMPOSING A JERK LIMIT IS ADEQUATE TO CONTROL STRUCTURAL EXCITATION. HOWEVER, WHEN THE STRUCTURAL FREQUENCY IS LOW, APPROXIMATELY 1 HZ, THE JERK AND ACCELERATION LIMITS ALONE ARE NOT ADEQUATE. THIS VIRTUAL TARGETING APPROACH CAN STILL RESULT IN LONG SYSTEM SETTLING TIMES FOR CERTAIN MANEUVER ANGLES.

Low Frequency Structure Results

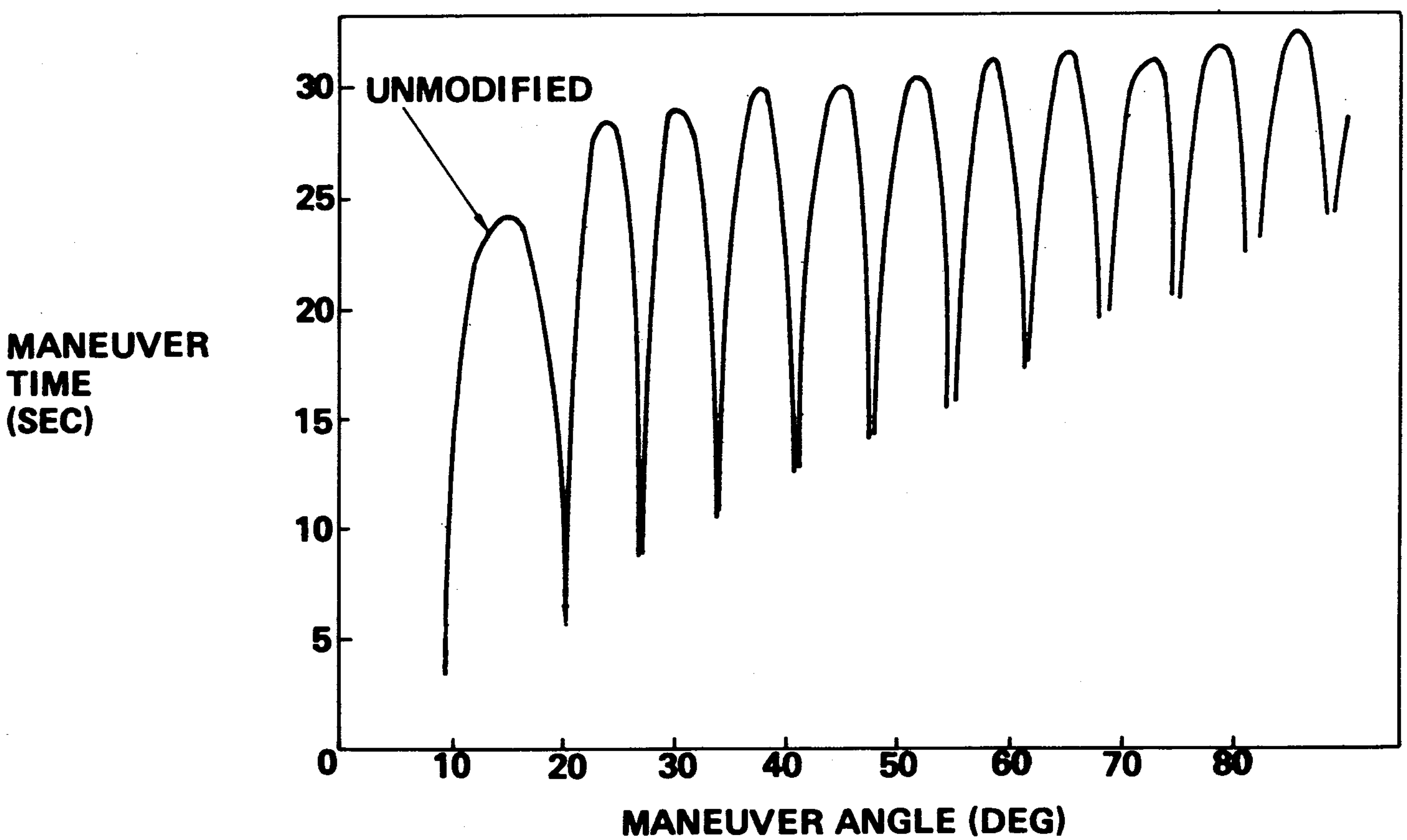

Figure 5

CONTROL SYSTEM DESIGN PROBLEM (Figure 6)

AN ALTERNATE APPROACH TO THE SETTLING TIME WOULD BE TO ADD EFFECTIVE DAMPING TO THE STRUCTURAL MODES USING THE CONTROL SYSTEM. HOWEVER, AS THE STRUCTURAL FREQUENCIES DROP, IT BECOMES VERY DIFFICULT TO ADD DAMPING TO THE STRUCTURAL MODE WITHOUT SEVERLY COMPROMISING THE RIGID BODY CONTROL FREQUENCY. THIS IS ILLUSTRATED BY THE ROOT LOCUS TRAJECTORIES SHOWN IN THE FIGURE.

Control System Design Problem

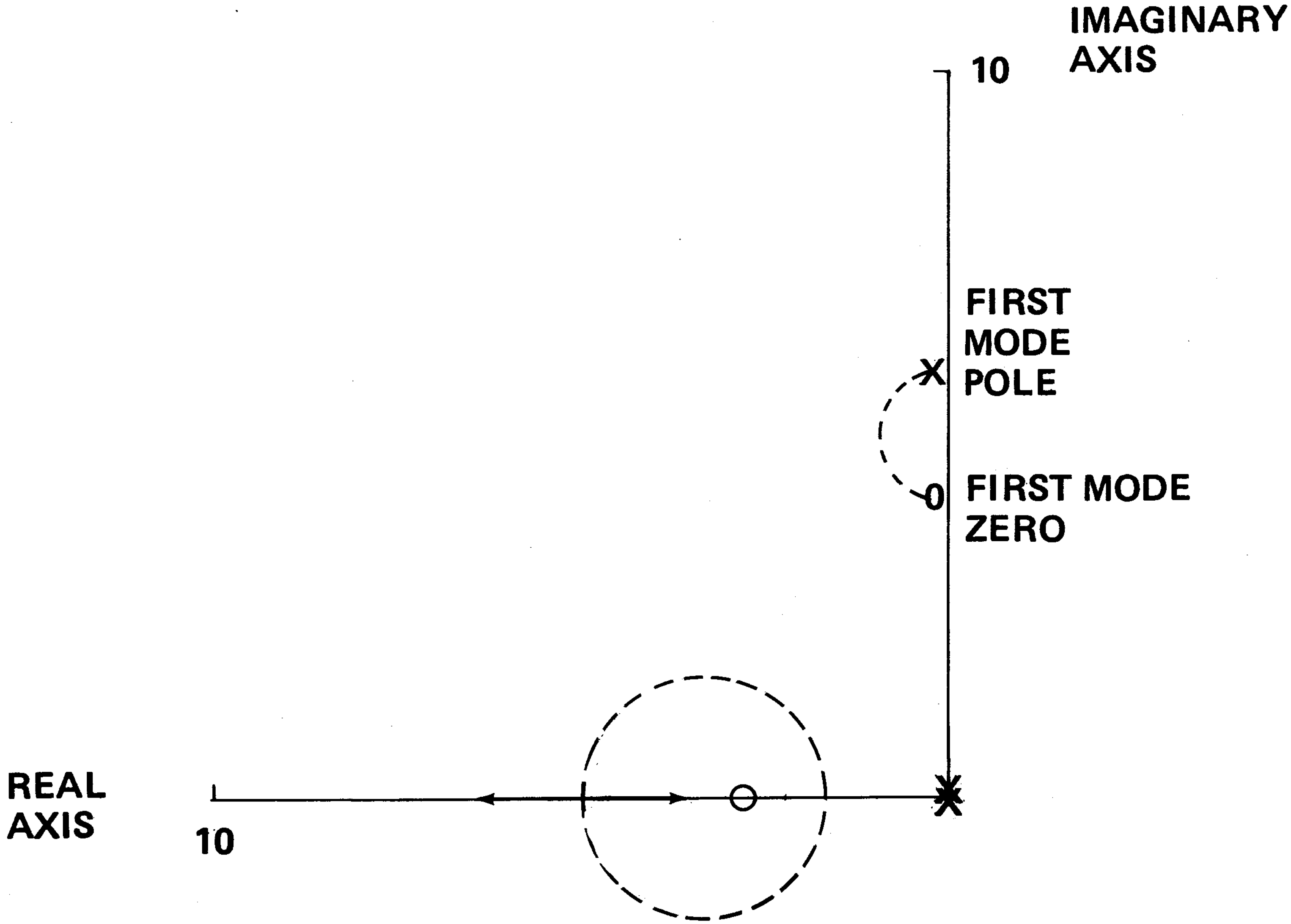

Figure 6

TEXT

LOW FREQUENCY STRUCTURE ANSWER (Figure 7)

FOR THE LOW FREQUENCY STRUCTURE, A NEW APPROACH HAS BEEN DEVELOPED WHICH CORRELATES THE LENGTH OF THE JERK IMPULSE WITH THE DOMINANT STRUCTURAL FREQUENCIES. THE SINUSOIDAL SHAPE IN THE FIGURE REPRESENTS THE STRUCTURAL MOTION CAUSED BY THE JERK IMPULSES. FOR THE UPPER FIGURE, THE STRUCTURAL FREQUENCY IS PROPERLY PHASED WITH THE JERK IMPULSES. IN THE LOWER FIGURE, THE TIMING IS COMPLETELY WRONG AND THE STRUCTURAL MOTION IS REINFORCED BY THE JERK IMPULSES. THE BENEFIT OF PROPER TIME PHASING IS OBVIOUS.

Low Frequency Structure Answer

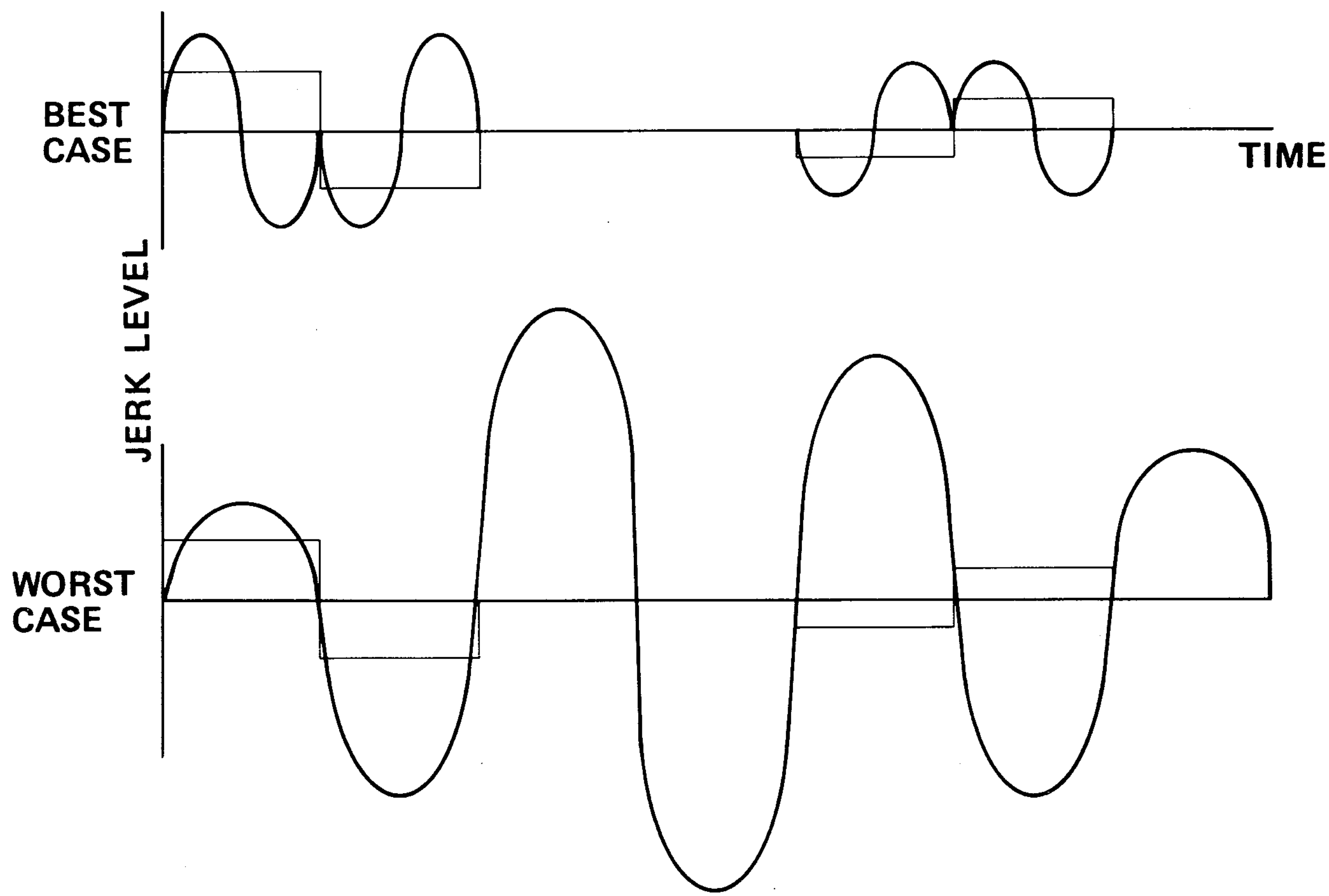

Figure 7

TEXT

ANALYTICAL SIMULATION RESPONSE (Figure 8, Figure 9)

THE NEXT TWO CHARTS SHOW THE RESULTS FROM AN ANALYTICAL SIMULATION USING THE NEW LOW FREQUENCY SLEW LAW. THE DESIGN GOAL FOR THIS SYSTEM WAS TO HAVE LESS THAN 100 MICRORADIANS OF ERROR ON THE ROLL AND PITCH AXIS LESS THAN 1.5 SECONDS AFTER THE END OF THE MANEUVER. THE DATA SHOWS THAT THIS OBJECTIVE WAS MET. THE DOMINANT MODE IN THIS SYSTEM WAS AT 1.1 Hz.

Analytical Simulation Response

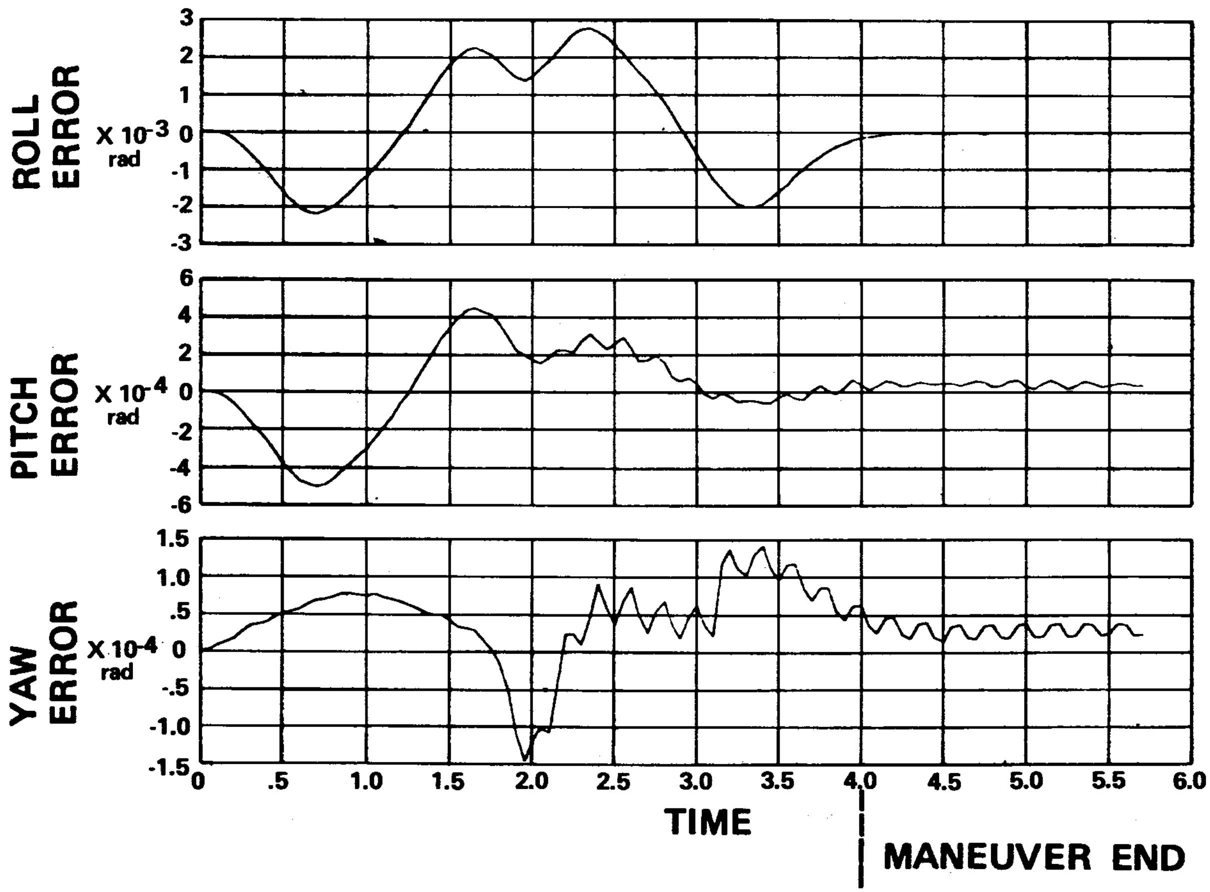

Figure 8

Analytical Simulation Response

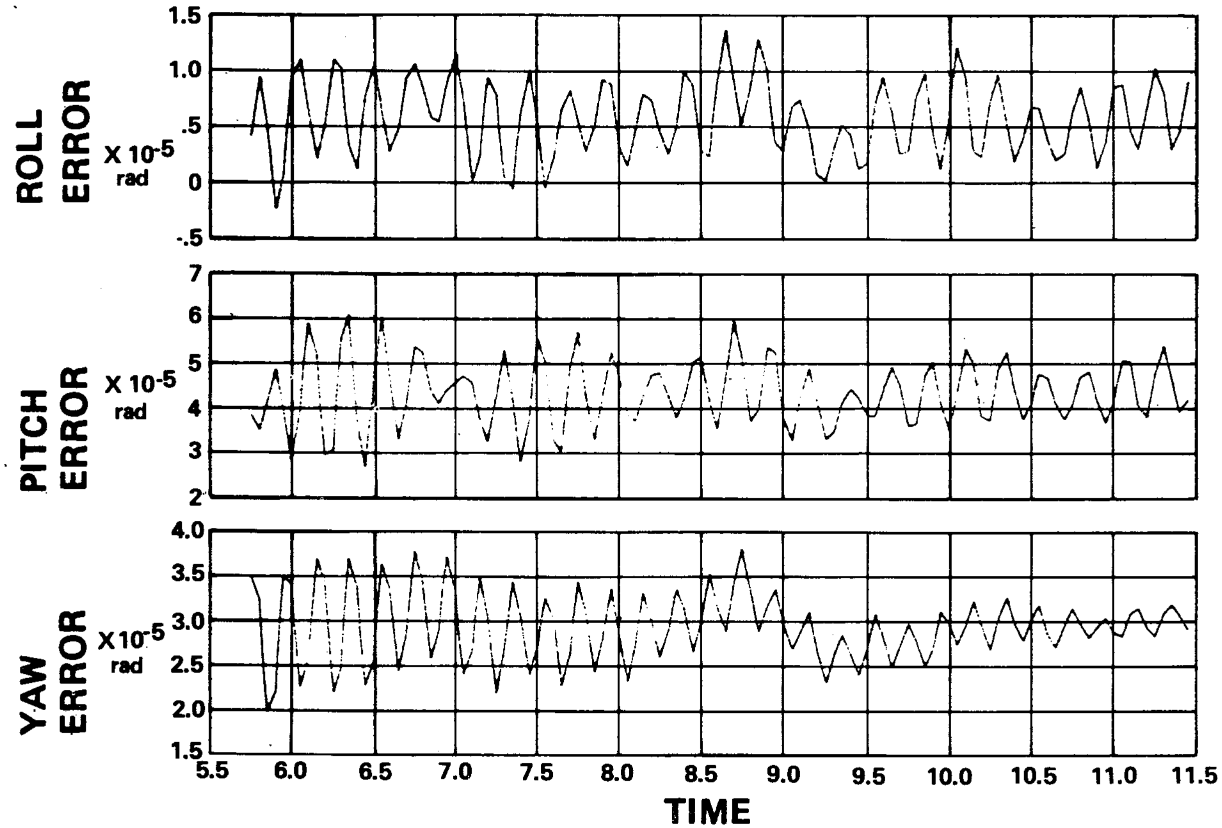

Figure 9

1008 TEXT

SIMULATION FUNCTIONAL DIAGRAM (Figure 10)

FINAL PROOF OF THE SYSTEM CONCEPT WAS VERIFIED USING A SCALE MODEL AIRBEARING SIMULATOR. THIS HARDWARE WAS CONFIGURED AS SHOWN HERE IN A FLIGHT — TYPE OF CONFIGURATION AND REQUIRED GENERATION OF FLIGHT-TYPE OPERATIONS SOFTWARE TO CONTROL THE SYSTEM IN THE REAL TIME ENVIRONMENT. THE CHARACTERISTICS OVER-ALL ARE SIMILAR TO THAT WHICH WOULD BE OBTAINED WITH ACTUAL FLIGHT-TYPE HARDWARE.

Simulation Functional Diagram

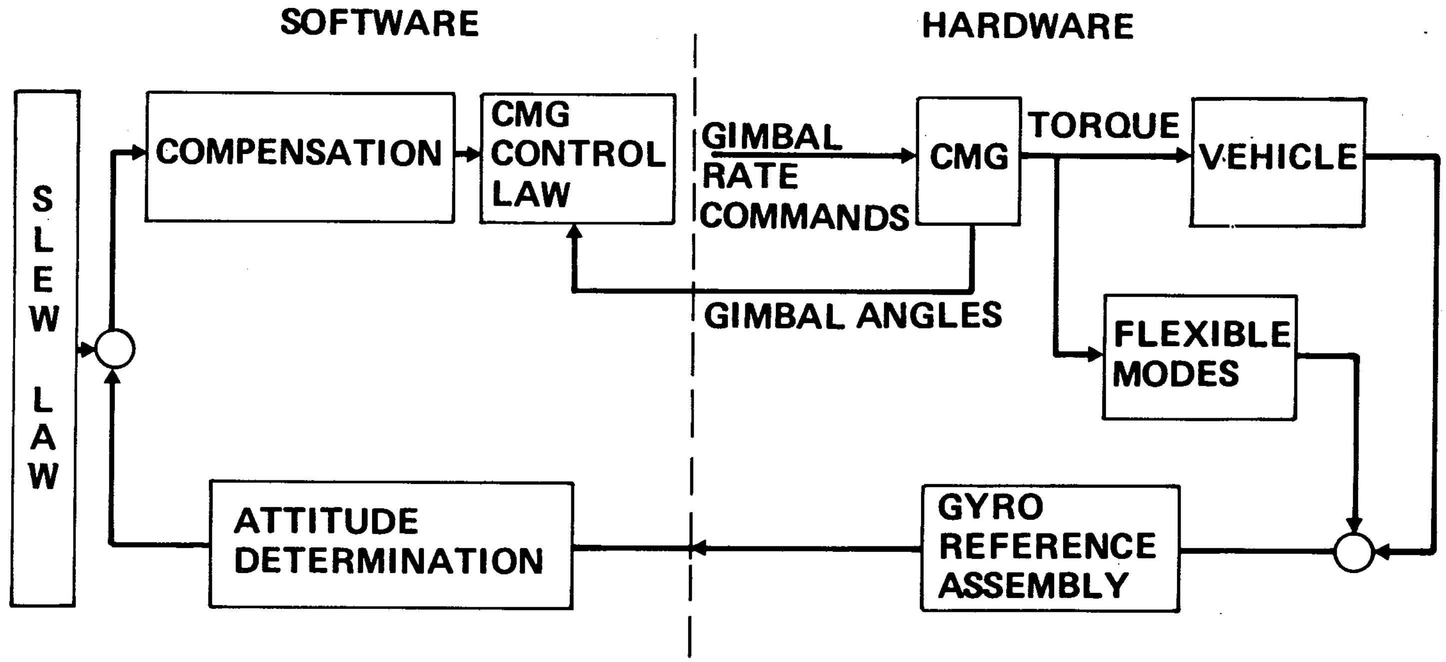

Figure 10

TEXT

ACTUATOR CONFIGURATION (Figure 11)

THIS FIGURE SHOWS SCHEMATICALLY THE SKEWED CMG CONFIGURATION USED FOR THE TEST: THIS CONFIGURATION WAS USED TO ASSURE THAT THE TEST RESULTS WOULD BE REALISTIC AND EXHIBIT THE SAME TYPE OF CROSS-COUPLED OPERATION THAT WOULD BE EXPERIENCED WITH A REAL SPACECRAFT.

Actuator Configuration

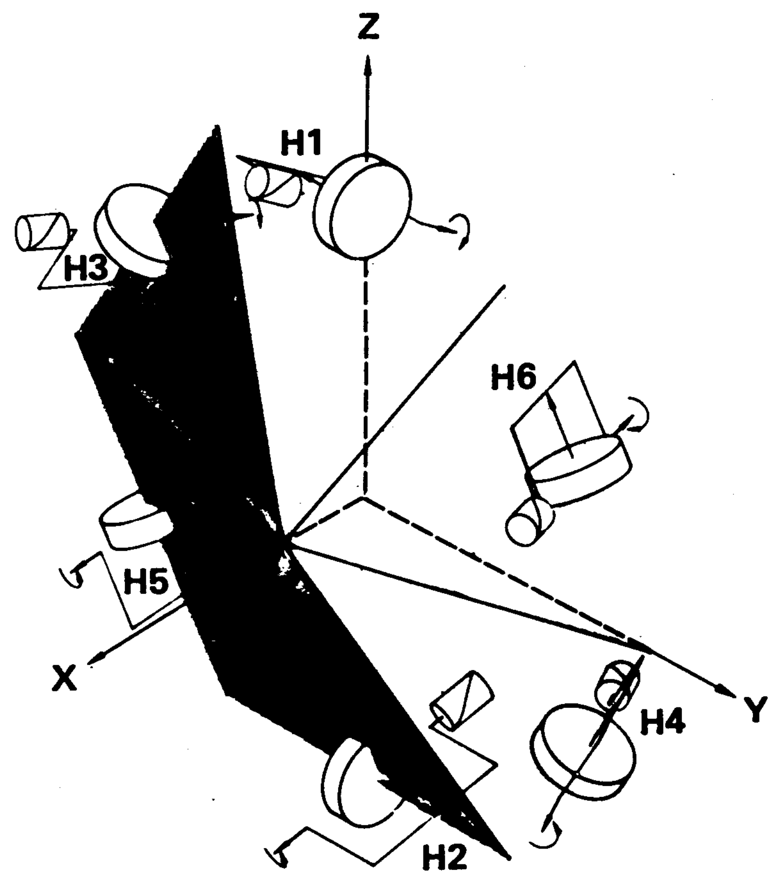

Figure 11

TEXT

HARDWARE SIMULATION (Figure 12)

THIS PICTURE SHOWS THE SCALE MODEL THREE AXIS AIRBEARING SIMULATOR. IT CONSISTS OF 2 SEPARATE SEGMENTS — A CENTRAL CORE WITH THE OPERATING HARDWARE ON IT, AND A PIPE FRAME WHICH IS USED TO REPRESENT THE FLEXIBLE VEHICLE CHARACTERISTICS. THE CENTRAL CORE CONTAINS THE 6 CMG's, COMPUTER AND GYRO REFERENCE ASSEMBLY.

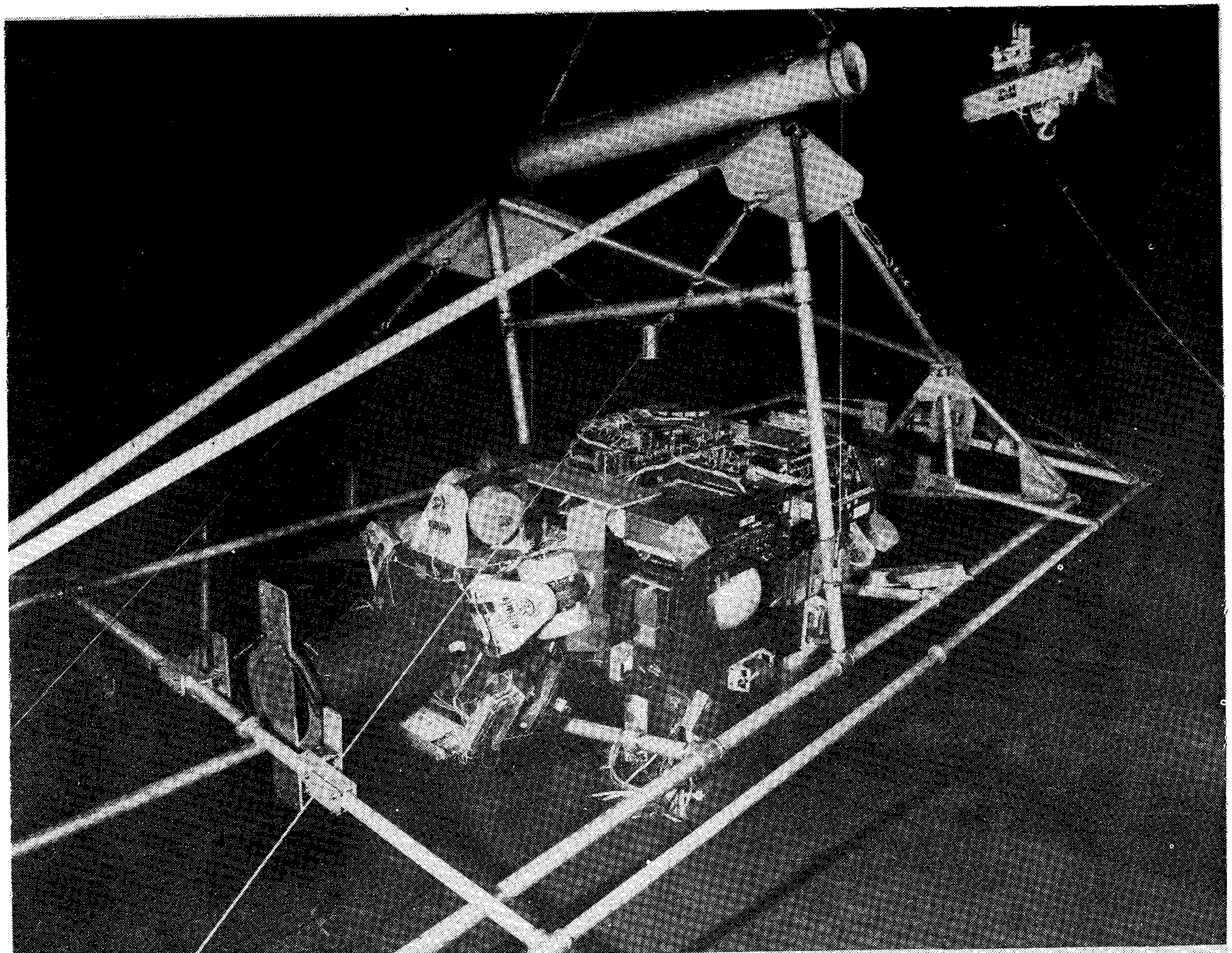

Figure 12

FLEXIBLE MODE COUPLING (Figure 13)

THIS FIGURE SHOWS, FOR A SINGLE AXIS, THE METHOD WHICH HAS BEEN DEVELOPED TO SIMULATE THE FLEXIBLE MODE CHARACTERISTICS. IN ORDER TO SIMULATE STRUCTURAL FREQUENCIES AS LOW AS 0.5 HZ, ADDITIONAL SUPPORT METHODS MUST BE USED SEPARATE FROM THE MAIN AIRBEARING SYSTEM. IF THIS IS NOT DONE, VEHICLE MOTIONS AWAY FROM A NORMAL LEVEL ORIENTATION WILL BE STRONGLY INFLUENCED BY SYSTEM UNBALANCE. THIS DISTURBANCE MASKS THE DESIRABLE OPERATING CHARACTERISTICS. BY USING A SUPPORT FROM THE CEILING, THE FLEXIBLE MODE CAN BE SIMULATED BY USING A SIMPLE SPRING CONNECTION BETWEEN THE OUTER FRAME AND THE CORE SIMULATOR. THE NORMAL SYSTEM SHOWN IN THE PRECEDING PICTURE USES A 3-AXIS GIMBAL SYSTEM TO SUPPORT THE PIPE FRAME WORK AND ALLOW IT TO MOVE ABOUT ALL THREE AXIS IN COORDINATION WITH THE CORE SIMULATOR.

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Flexible Mode Coupling

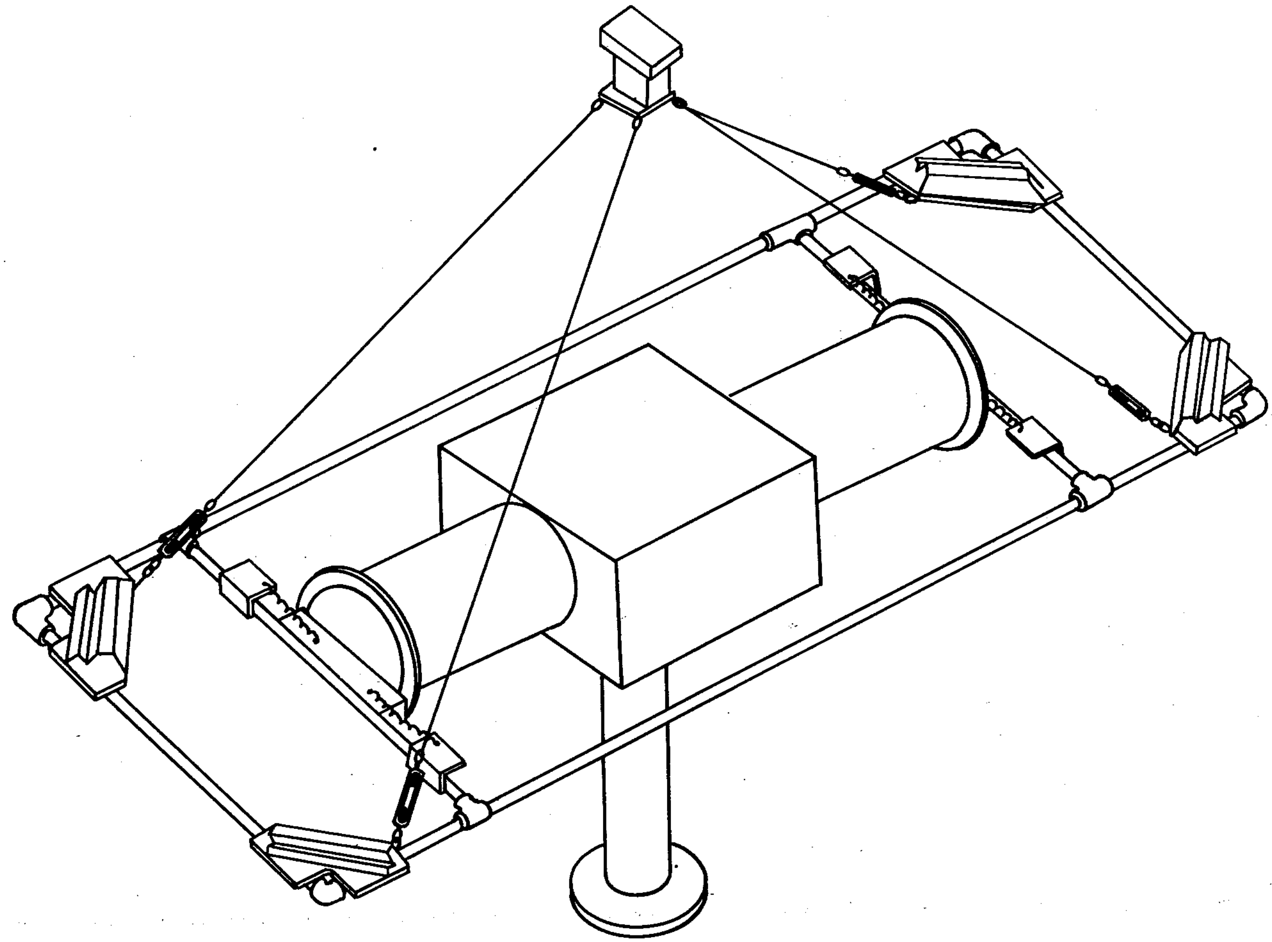


Figure 13

TEST RESULTS (Figure 14)

THIS HARDWARE RESULTS CHART SHOWS A COMPARISON BETWEEN THE OLD AND NEW VIRTUAL TARGET SLEW LAW APPROACHES. THE NEW APPROACH DOES IMPOSE A SMALL RIGID BODY MANEUVER TIME PENALTY; BUT THE SETTLING TIME FOR THE NEW APPROACH IS REDUCED FROM 24 SECONDS WITH THE OLD APPROACH TO LESS THAN 1 SECOND WITH THE NEW APPROACH. THE TOTAL TIME FOR THE NEW APPROACH, INCLUDING SETTLING TIME, IS SUBSTANTIALLY LESS THAN THE OLD JERK LIMITED SLEW LAW.

Test Results

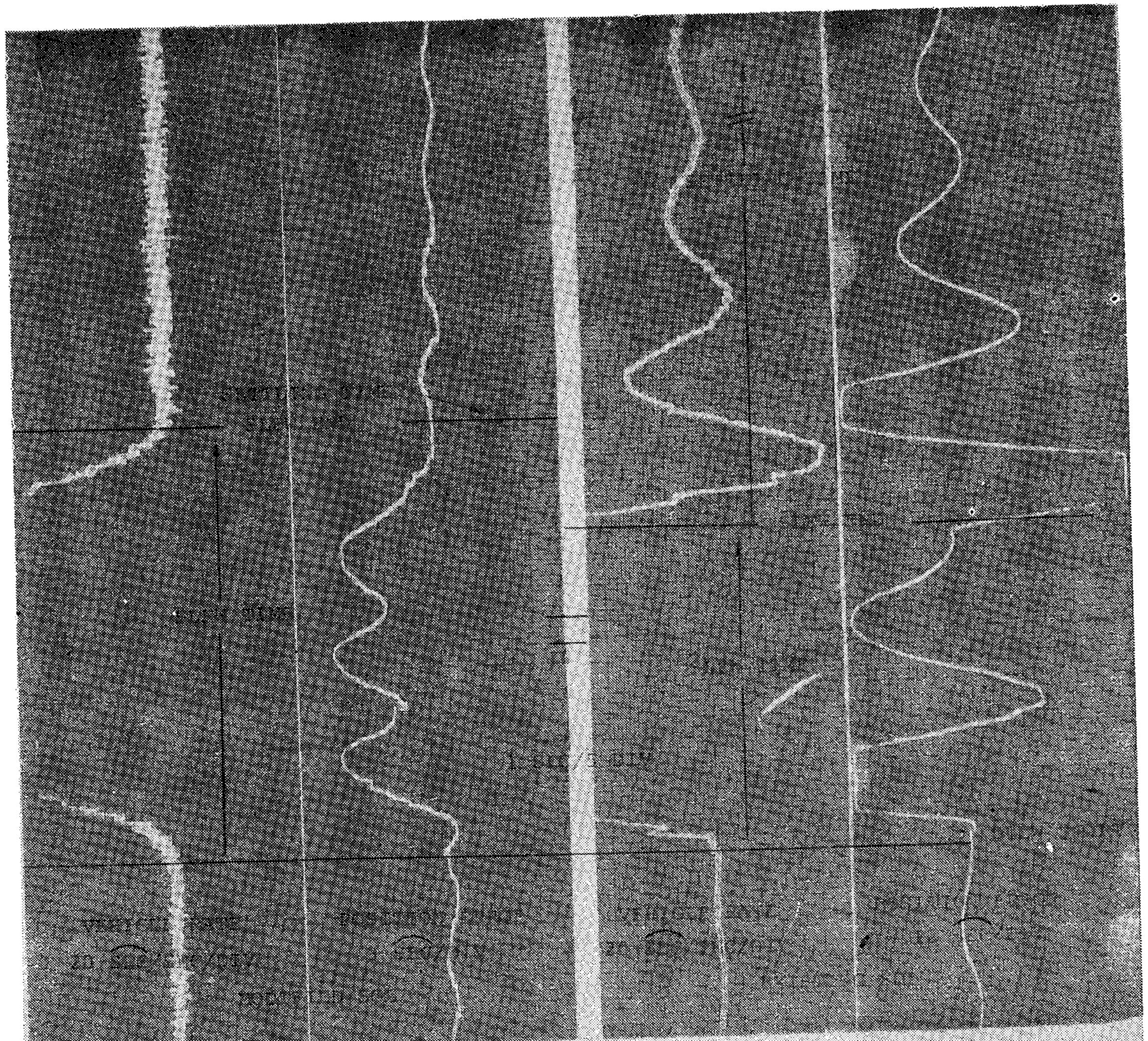


Figure 14

TEXT

NEW SIMULATOR (Figure 15)

In 1977, a significant modification was made to this simulation facility. 4 new CMG's were purchased from Sperry, Inc. and installed on this simulator. This new hardware will allow the vehicle to maneuver at rates up to 8 degrees per second. This is an increase by a factor of 10 over that achievable with that of the other hardware when the flexible appendage was attached.

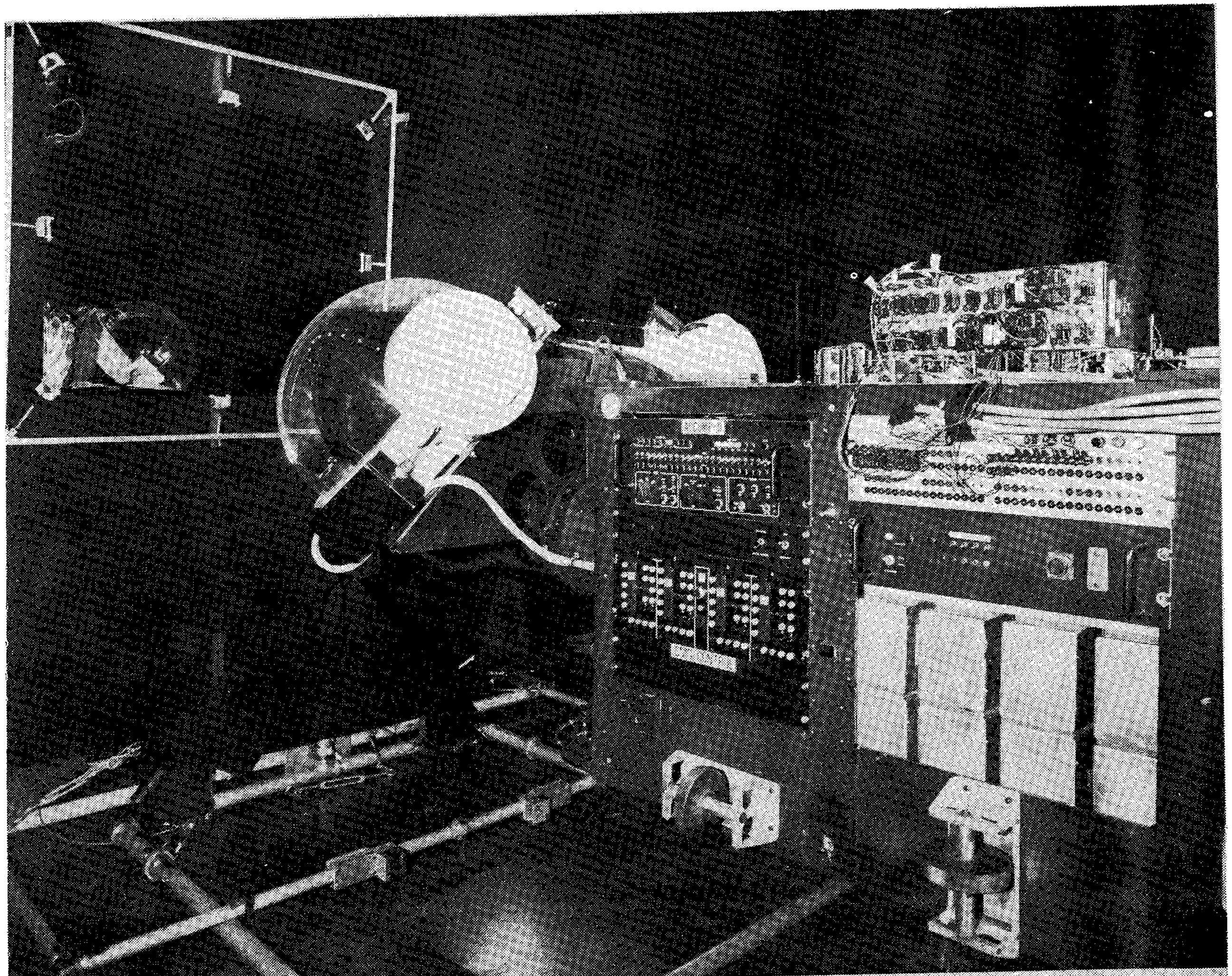


Figure 15

Summary

- MANEUVER AND POINTING OF FLEXIBLE VEHICLES SOLVED FOR SEVERAL CLASSES OF VEHICLE
- SLEW LAW WORKS AS EXPECTED
- ANALYSIS TOOLS ARE AVAILABLE
- GOOD CORRELATION OBTAINED BETWEEN ANALYSIS AND TEST
- VERY LOW FREQUENCY STRUCTURE IS NEXT PROBLEM AREA

Figure 16

THE PRECISION SELF-METERING STRUCTURE (PSMS)

BY

W.C. YAGER

FOR PRESENTATION AT THE

NASA/DOD/INDUSTRY SEMINAR ON
LARGE SPACE SYSTEMS TECHNOLOGY

LANGLEY RESEARCH CENTER

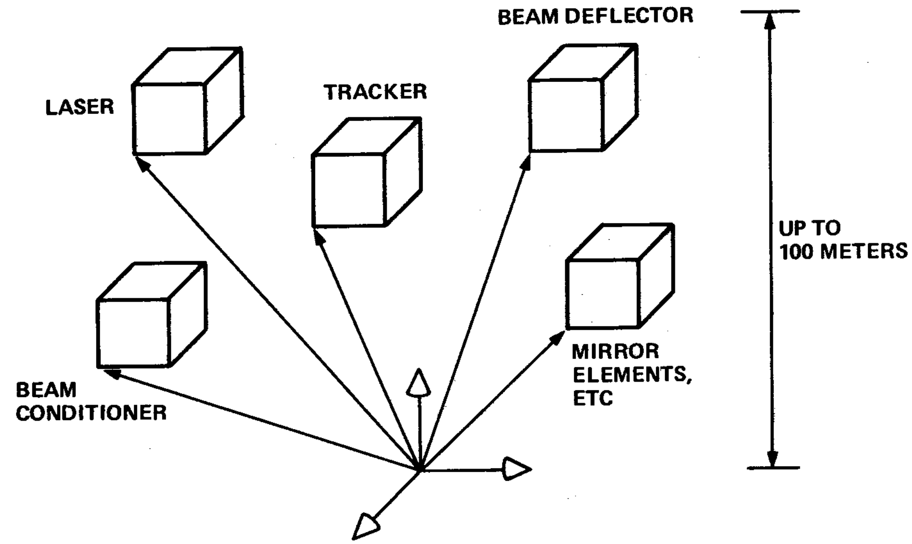
JANUARY 17-19, 1978

GENERAL ELECTRIC
Re-entry & Environmental Systems Division
PHILADELPHIA, PENNSYLVANIA

ORIGIN AND SCOPE OF THE PROBLEM (Figure 1)

Large, precise space systems such as space lasers, space telescopes, and space power transmitters cannot be realized until certain fundamental metrological problems are first solved. It must be shown (1) how a spatially distributed system of elements can be tied together in terms of a master coordinate system, (2) how master coordinates for these distributed elements can be determined with great accuracy, and (3) how mechanical integration of the elements to desired master coordinates of such accuracy can be achieved.

ORIGIN AND SCOPE OF THE PROBLEM



FUNCTION REQUIRES GEOMETRICAL INTEGRATION TO 10^{-6}m, 10^{-7} rad
BUT
SYSTEM ELEMENTS ARE PHYSICALLY SEPARATED BY UP TO 100 m

METROLOGY	COORDINATE INTEGRATION	PUT SPATIALLY DISPERSED SYSTEM OF ELEMENTS ON MASTER COORDINATE SYSTEM
	COORDINATE DETERMINATION	DETERMINE MASTER COORDINATES OF ELEMENTS IN REAL TIME TO 10^{-6} m, 10^{-7} rad
STRUCTURE & CONTROLS	MECHANICAL INTEGRATION	ACHIEVE AND MAINTAIN DESIGN MASTER COORDINATES TO 10^{-6}m, 10^{-7} rad

Figure 1

MORPHOLOGY OF METRICAL STRUCTURES (Figure 2)

Three fundamental operations are available to the metrology designer: spacing, alignment, and collimation. Assuming at least one spacing operation is always necessary to establish scale, four general types of metrology systems are possible. The PSMS metrology system is based on the use of spacing operations only.

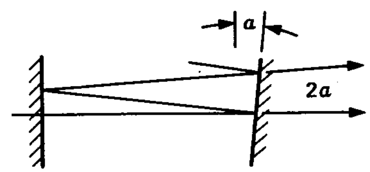
MORPHOLOGY OF METRICAL STRUCTURES

BASIC METRICAL OPERATIONS

1. SPACING

2. ALIGNMENT

3. COLLIMATION



TYPES OF METRICAL SYSTEMS

TYPE 1: SPACING ONLY

TYPE 2: SPACING AND ALIGNMENT

TYPE 3: SPACING AND COLLIMATION

TAPE 4: SPACING AND ALIGNMENT AND COLLIMATION

THE PSMS SYSTEM

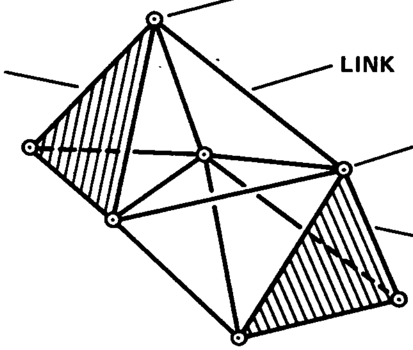
A TYPE 1 SYSTEM

Figure 2

A SPACER-ONLY INTEGRATION CONCEPT (Figure 3)

Spacer-only integration can be realized in a spaceframe of the ball-strut type. In the PSMS system, the load-bearing and dimensional control functions are each assigned to a separate ball-strut spaceframe; these two structures are then arranged to occupy the same space, that is, they are made coaxial. Load-bearing is handled by a hollow, ball-strut exostructure. Dimensional control is handled by a metrical endostructure.

A SPACER-ONLY INTEGRATION CONCEPT



CONCEPT

1. DEFINE PLATFORMS IN TERMS OF NODES
2. INTERPOSE CONNECTING NODES AND LINKS
3. CONTINUOUSLY DETERMINE LENGTHS OF ALL LINKS
4. FROM LINK LENGTHS COMPUTE NODE COORDINATES
5. ADJUST NODES TO ACHIEVE DESIRED COORDINATES

IMPLEMENTATION

1. BASIC DESIGN: BALL-STRUT SPACEFRAME
2. LOAD- BEARING FUNCTION: EXOSTRUCTURE
3. DIMENSIONAL CONTROL FUNCTION: ENDOSTRUCTURE

Figure 3

LOAD-BEARING EXOSTRUCTURE (Figure 4)

The load-bearing exostructure will have the general form of hollow balls at the nodes, with hollow struts connecting the nodes. Many mechanical design variations are possible, but, in general, a telescoping feature and a spherical seating feature should be included among the strut details to facilitate erection.

LOAD-BEARING EXOSTRUCTURE

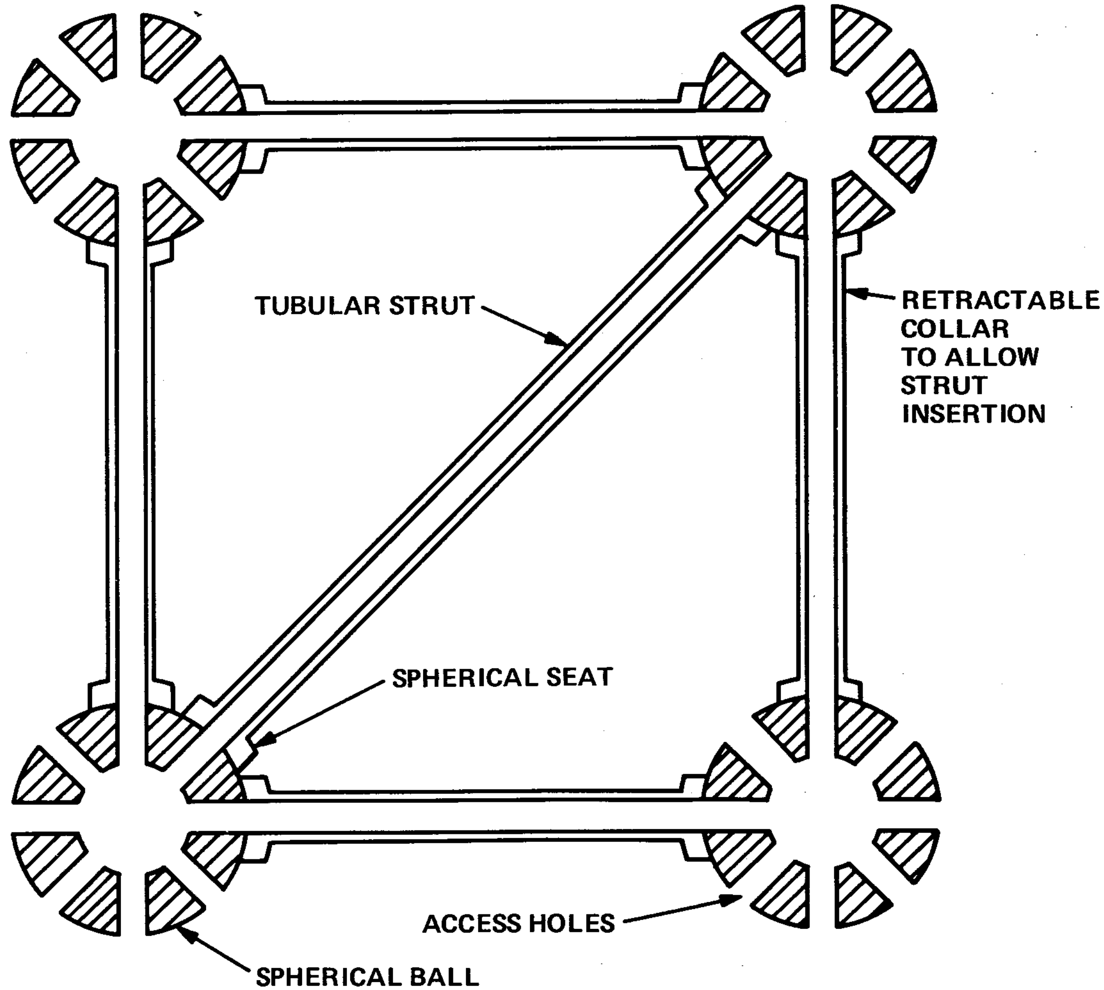


Figure 4

METRICAL ENDOSTRUCTURE (Figure 5)

The metrical endostructure will include (1) precision balls, whose surface centroids define the nodes, and (2) spring-loaded quartz spacer assemblies that bear against the nodal balls and complete the physical internodal link. A gas spring is preferred because it eliminates the need to calibrate for contact force. The spacer assembly will include at least two members (as a rod free to slide within a tube) arranged to allow a telescoping action. An absolute displacement gauge will indicate the relative movement of the two quartz spacers.

METRICAL ENDOSTRUCTURE

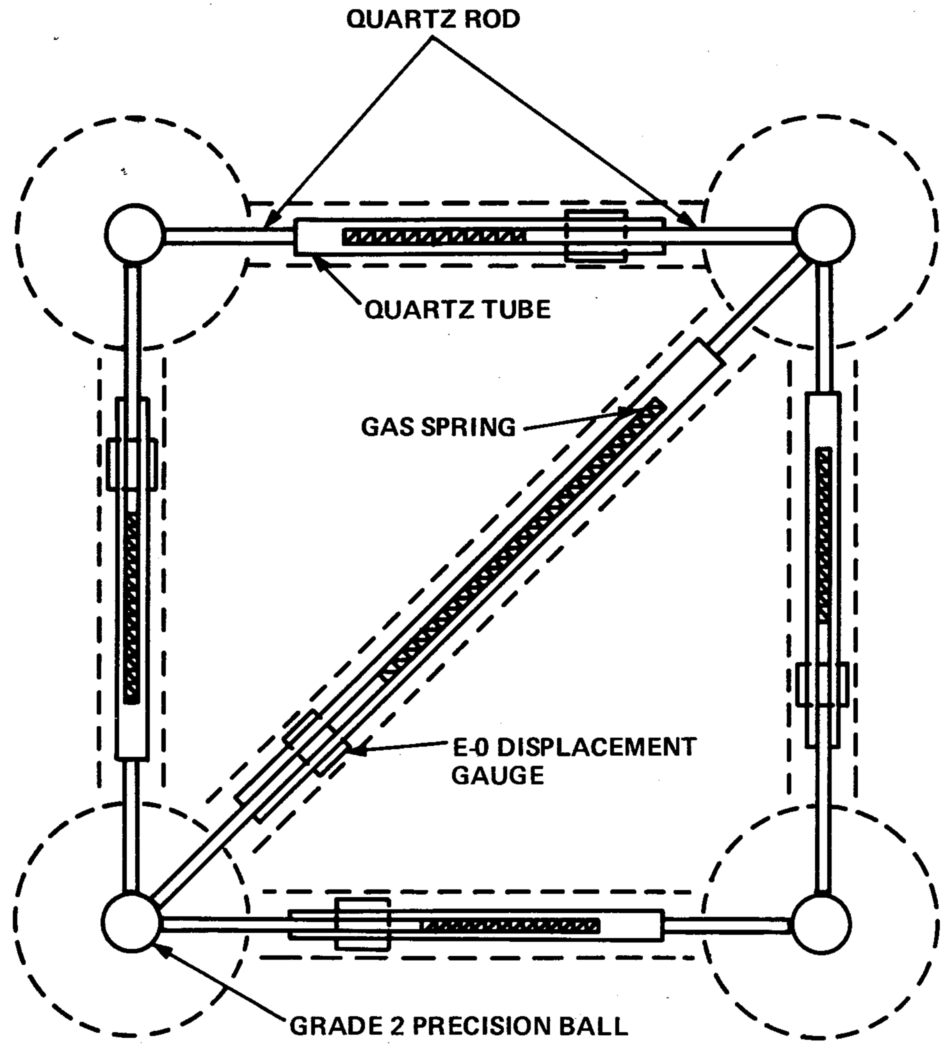


Figure 5

PLATFORM ATTACHMENT (Figure 6)

Three nodes can serve to define the reference plane of an attached platform. This reference plane can be transferred mechanically into the intra-platform alignment system by means of quartz spacers. Micropositioners developing reaction at the exostructure can move the reference nodes as required to meet coordinate control requirements. The endostructural spacer assemblies will follow the balls being moved to maintain the physical internodal link.

The endostructure interfaces with the exostructure primarily through the precision ball suspension. The precision ball can be suspended inside the hollow exostructural ball by a molded-in-place compliant buffer of RTV or similar material. This material will be molded so as to provide a receiving and guiding socket for the termination of the spring-loaded quartz spacer assembly.

PLATFORM ATTACHMENT

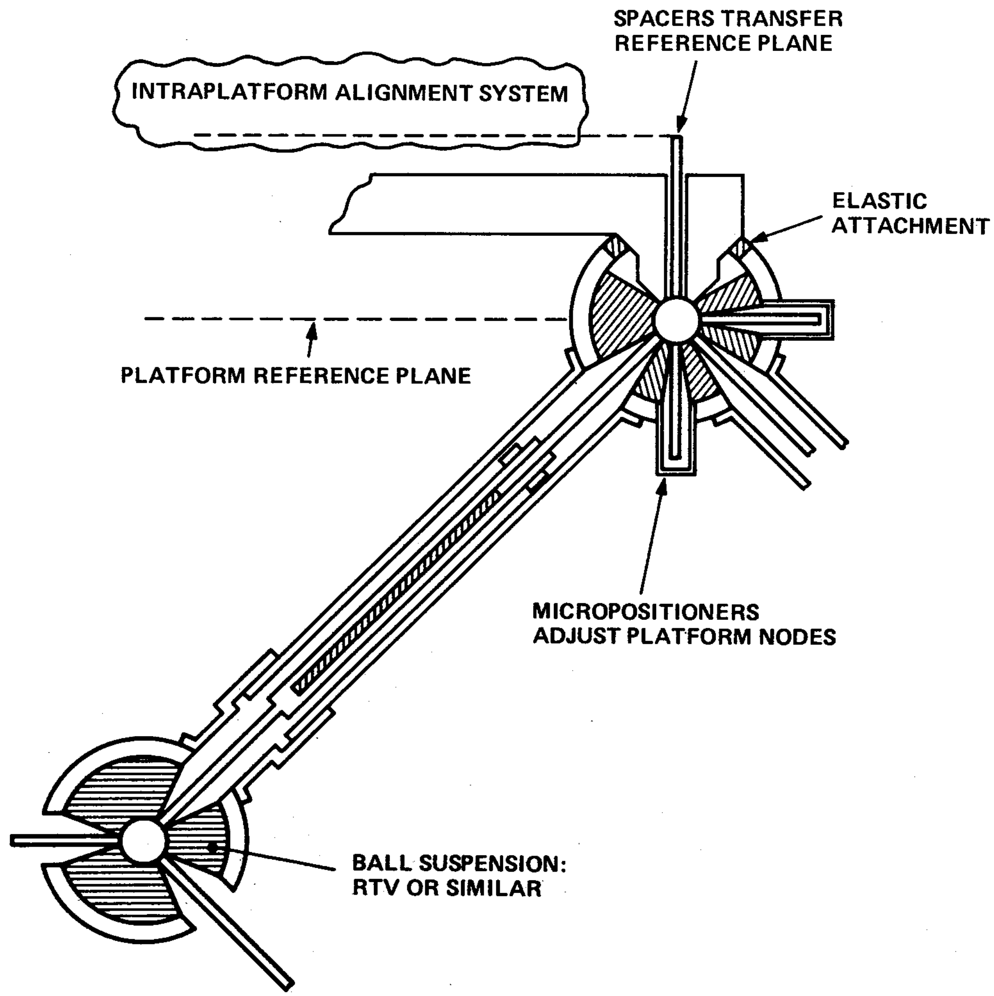


Figure 6

ANTECEDENTS: THE MERO SYSTEM (Figure 7; Figure 8)

A variety of ball-strut exostructural designs are compatible with the PSMS concept. One possibility is an adaptation of an old (circa 1940) German design called the MERO system.

ANTECEDENTS - THE MERO SYSTEM

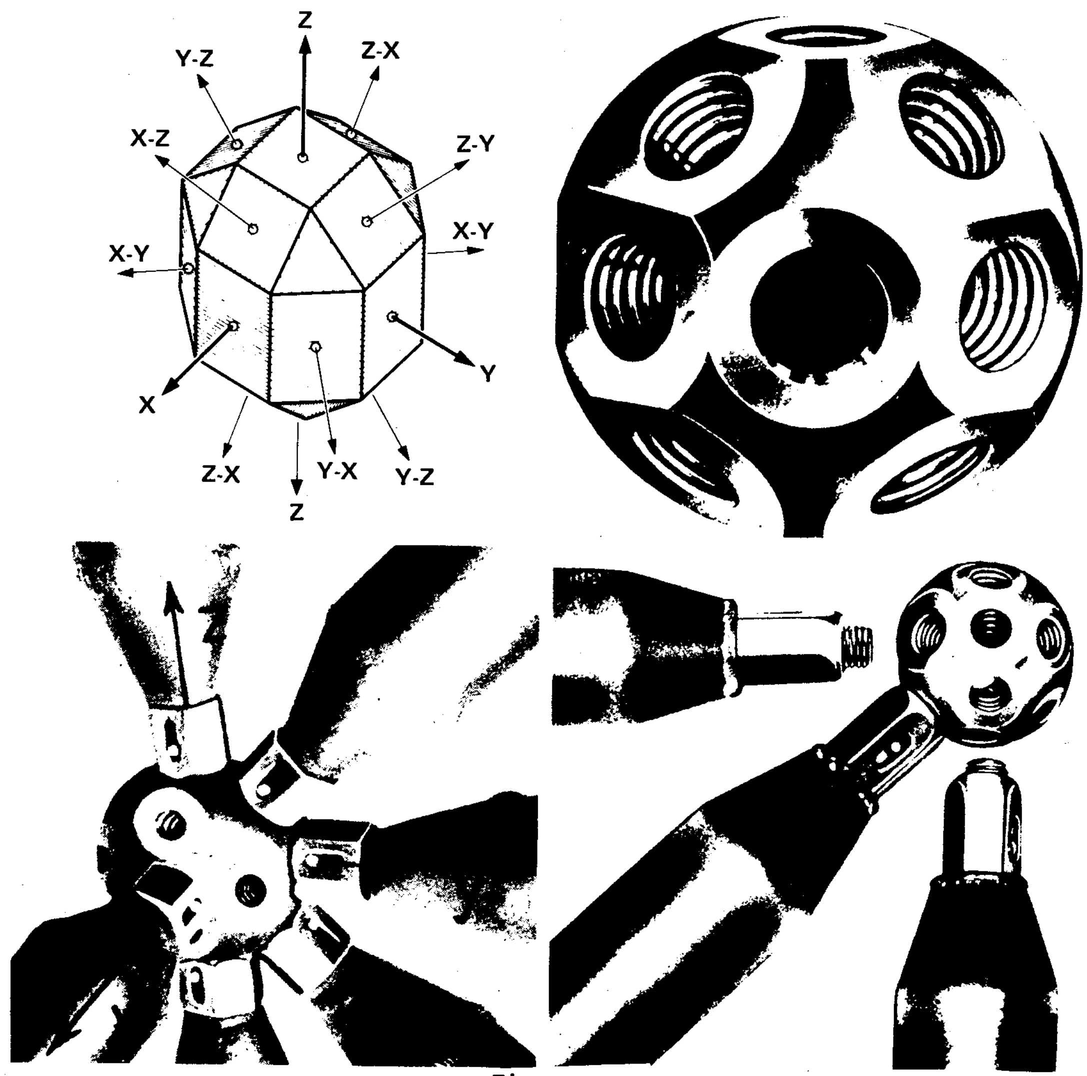


Figure 7

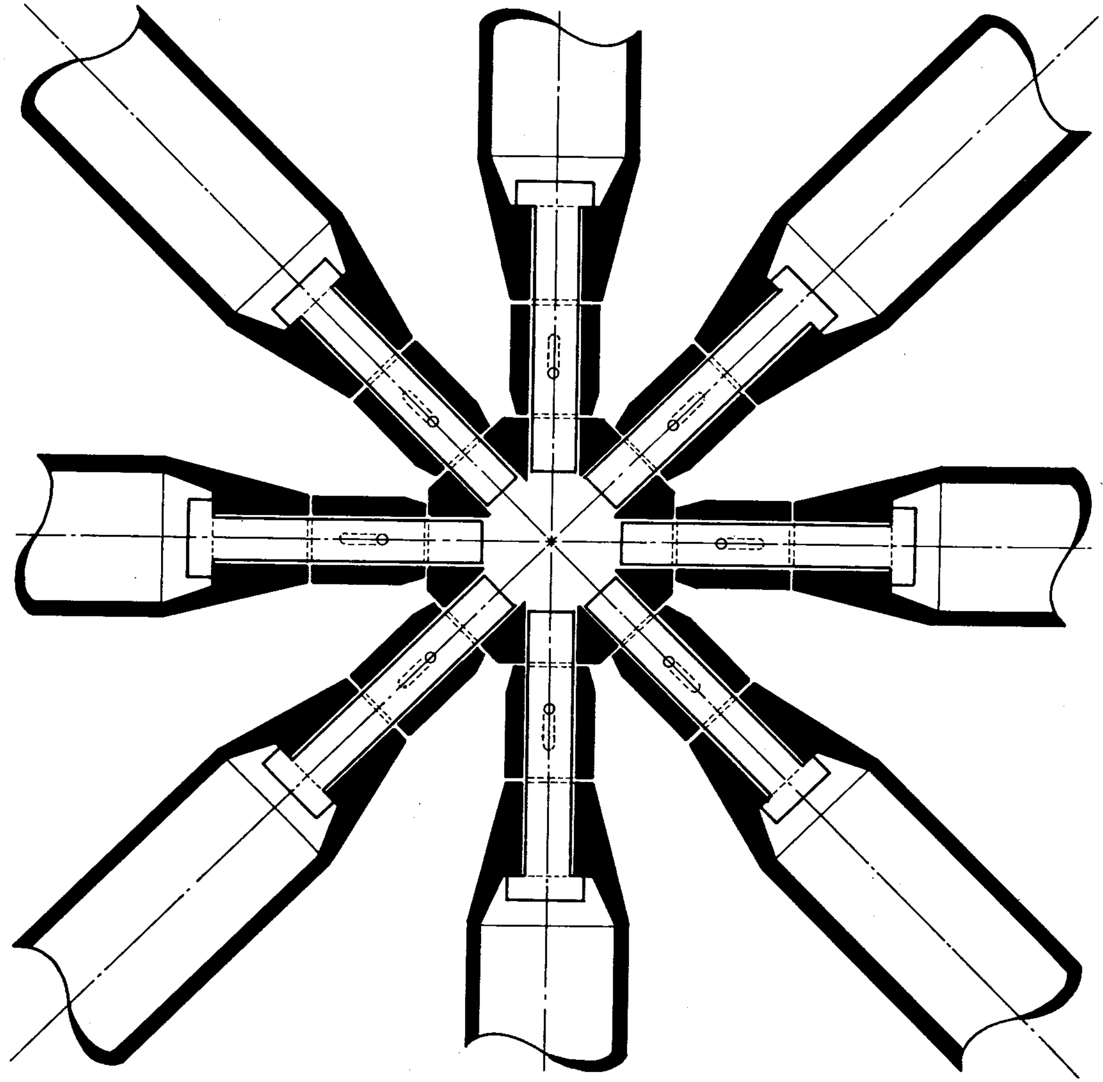


Figure 8

MODIFICATION OF THE MERO STRUT (Figure 9)

As it stands, the MERO design is not completely hollow and is severely overconstrained at the nodes. The joining bolt can easily be made hollow, however, and telescoping and spherical seating features can be added to improve erectability.

MODIFICATION OF THE MERO STRUT

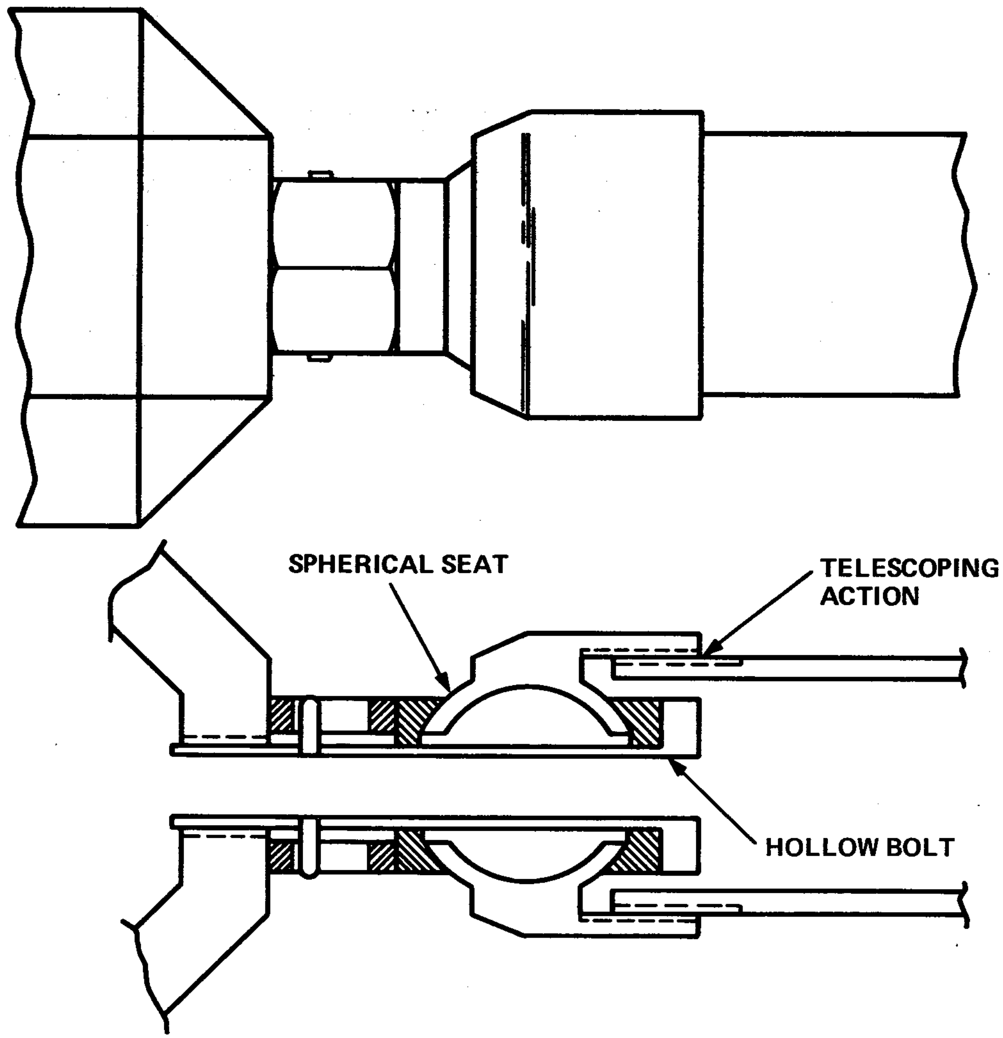


Figure 9

KEY ELEMENTS OF METRICAL ENDOSTRUCTURE (Figure 10)

The length attributed to any internodal link can be differentially affected by several processes, and each must therefore be either avoided or tracked to allow correction. Misalignment of spacer assembly axis and the line connecting the nodes is avoided by means of the socket or guideway molded into the ball suspension. Longitudinal movement of the balls is tracked by the absolute displacement indicator. Changes in the contact force compressing the spacers is tracked by a force transducer. (If the spring is pneumatic, force changes are functions of temperature, and the force transducer can be eliminated.) Temperature changes that alter the physical length of the spacers and balls are tracked by thermistors on all elements.

KEY ELEMENTS OF METRICAL ENDOSTRUCTURE

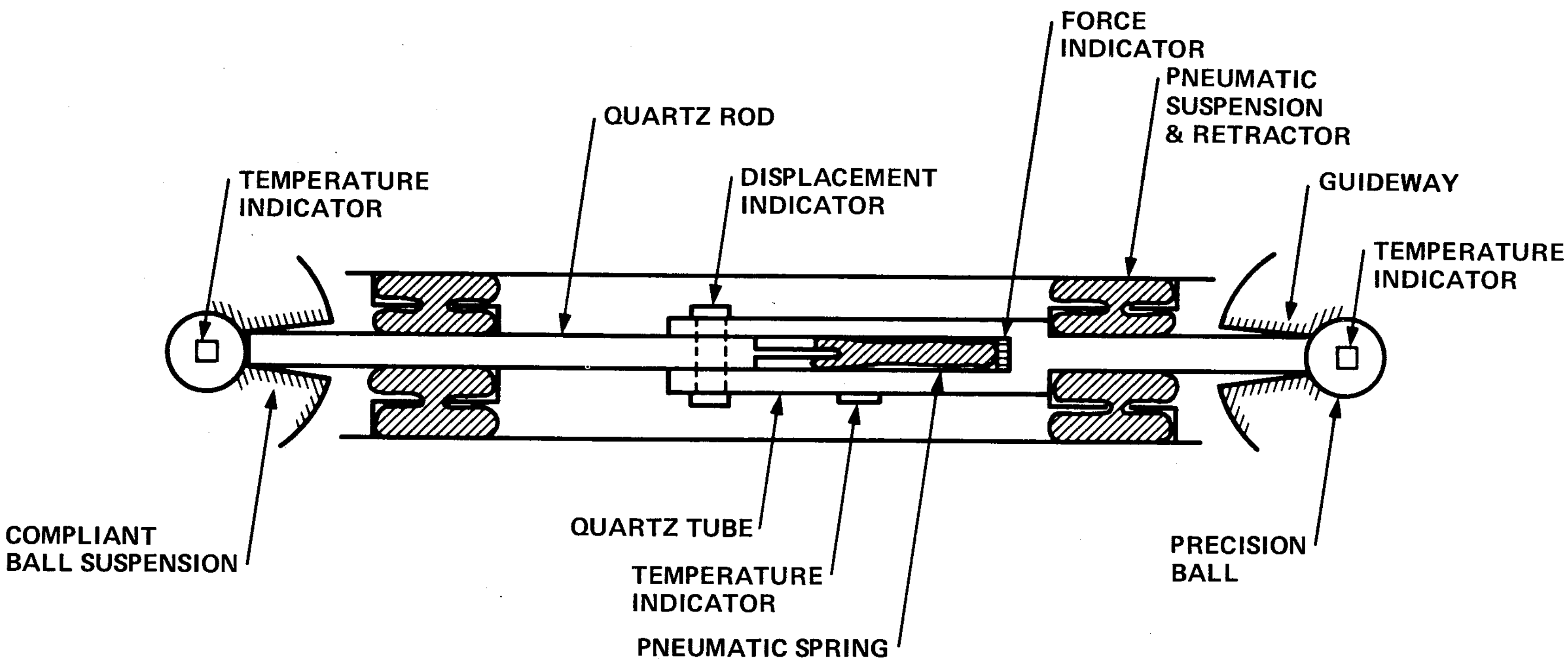


Figure 10

INSTALLATION SEQUENCE (Figure 11)

In the PSMS concept, the spacer assembly is calibrated while mounted in the strut, then retracted, and held pneumatically for storage and shipment. The metrology system is activated during erection simply by releasing the pneumatic suspension and making an electrical connection.

INSTALLATION SEQUENCE

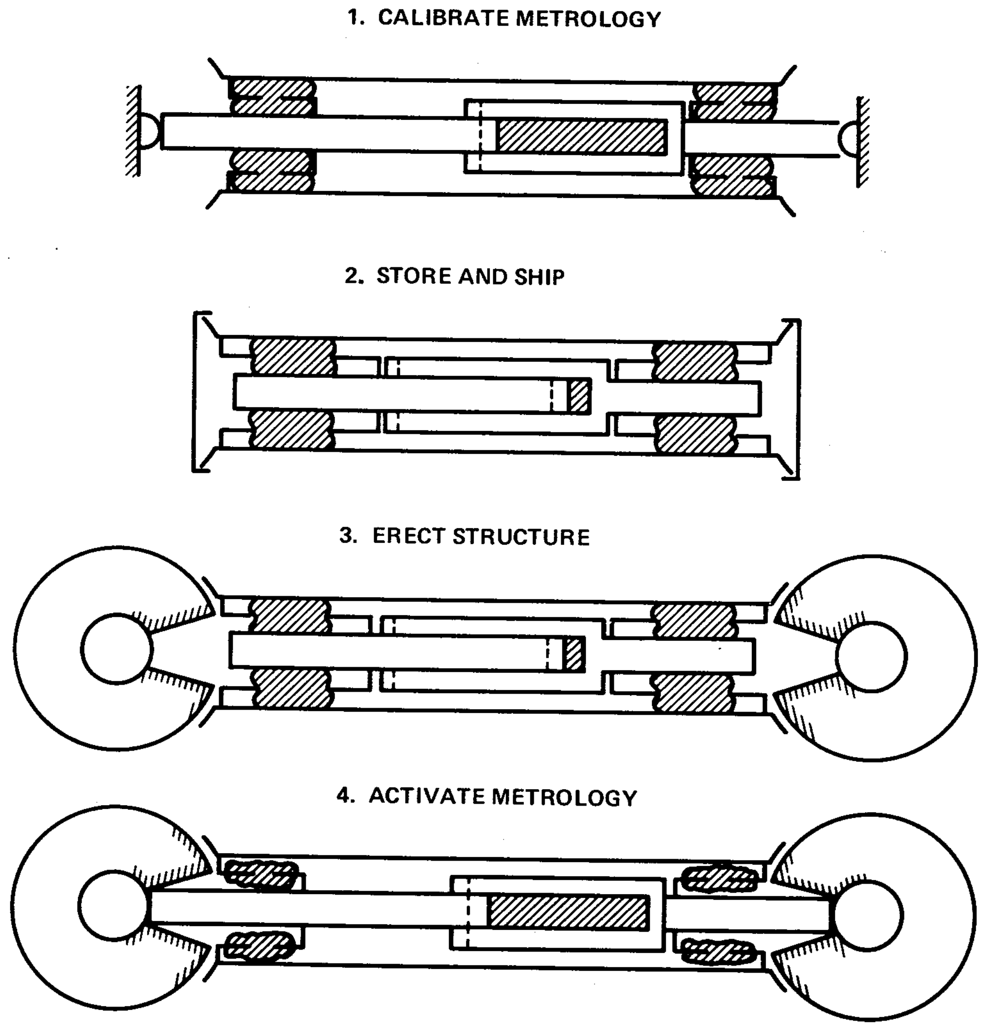


Figure 11

DIMENSIONAL CONTROL HIERARCHY (Figure 12)

In the PSMS concept all dimensional information is traceable to a local physical standard, a glass-ceramic bar. No reference is ever made to any outside measurement, calibration, or standard.

DIMENSIONAL CONTROL HIERARCHY

IN SPACE ← → IN GROUND LABORATORY

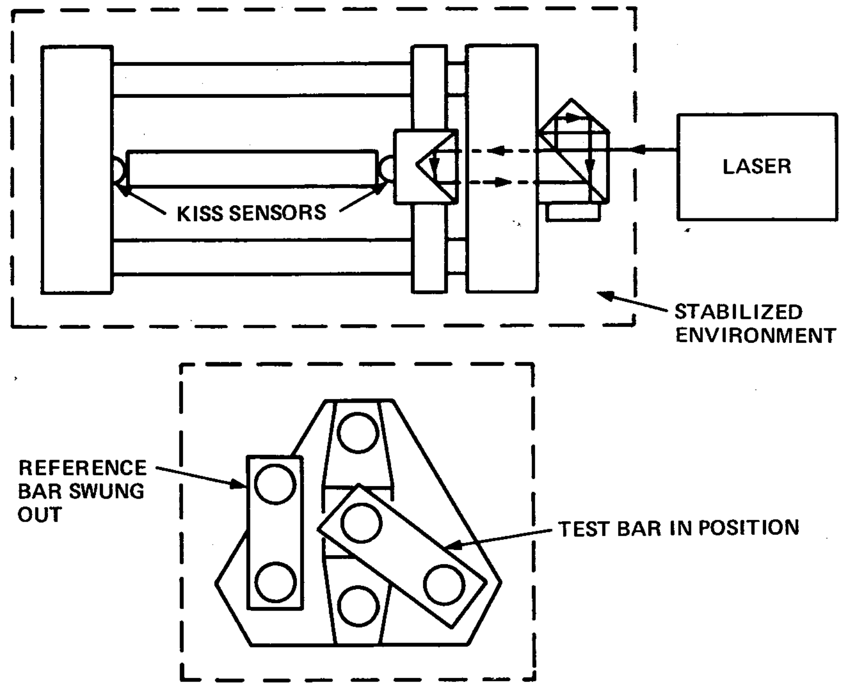
4th LEVEL ON-BOARD COMPUTATION	3d LEVEL ON-BOARD COMPUTATION	2d LEVEL ON-BOARD COMPUTATION	1st LEVEL ON-BOARD COMPUTATION	ON-BOARD INDICATOR READINGS	MAIN CALIBRATION	PRELIMINARY CALIBRATION	SECONDARY STANDARDS	PRIMARY STANDARD
POSITIONER COMMANDS	NODE COORDINATES	CURRENT LENGTH	LENGTH INCREMENTS	DISPLACEMENT, FORCE, TEMPERATURE	BASE LENGTH, CHANGE RATES	ON-BOARD INDICATORS	LABORATORY INSTRUMENTS	CERVIT OR ULE BAR
ΔX	X/S	L/S			$(L/S)_0$		$\lambda(10^{-8})$	S ($\approx$ 3m)
ΔY	Y/S							
ΔZ	Z/S		$\Delta(L/S)_D$	D(L)	$\alpha = \frac{\partial(L/S)}{\partial D(L)}$			
			$\Delta(L/S)_F$	F(L)	$\beta = \frac{\partial(L/S)}{\partial F(L)}$	F(L)		
			$\Delta(L/S)_T$	T(L)	$\gamma = \frac{\partial(L/S)}{\partial T(L)}$	T(L)	$F^{\#}(0.05)$	
						T(S)	T^{0} (0.05)	

Figure 12

BASELINE MEASUREMENT (Figure 13)

It is intended to use a commercial interferometer, the HP-5526A, for all comparisons, if the frequency stability proves adequate. Other lasers with adequate frequency stability can be substituted, if necessary.

BASELINE MEASUREMENT WITH HP-5526A LINEAR INTERFEROMETER



1. REFERENCE BAR OF CERVIT OR ULE APPROXIMATES LONGEST TEST BAR.
2. TEST BAR MEASUREMENTS ALTERNATE WITH REFERENCE BAR MEASUREMENTS.
3. LIGHTPATH INDEX STABILIZED TO $\approx 10^{-9}$ PER COMPARISON TIME.
4. LASER FREQUENCY STABILIZED TO $\approx 10^{-9}$ PER COMPARISON TIME.

Figure 13

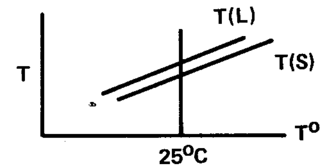
CALIBRATION PROCEDURES (Figure 14)

Calibration of the endostructure is done by interferometric comparison with a local standard under optimum and carefully controlled conditions in a ground laboratory. Baselengths and coefficients are determined from regression lines (or curves) based on many comparisons.

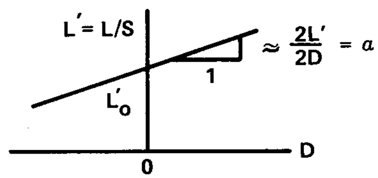
CALIBRATION PROCEDURES

1. PRELIMINARY

CORRELATE T(L) WITH T(S) IN TERMS OF $\hat{T}^{o}$(LAB), TO $0.05^{o}C$

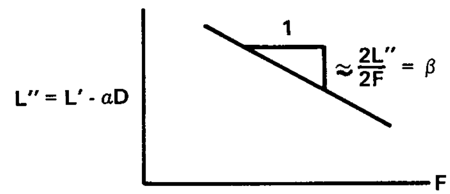


2. BASELENGTH AND DISPLACEMENT GAUGE



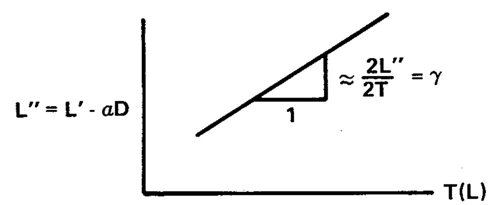
$T(L) \longrightarrow T^{o} = 25^{o}C$
$T(S) \longrightarrow T^{o} = 25^{o}C$
$F \longrightarrow 1$ LB

3. FORCE GAUGE



$T(L) \longrightarrow T^{o} = 25^{o}C$
$T(S) \longrightarrow T^{o} = 25^{o}C$

4. TEMPERATURE GAUGE



$T(S) \longrightarrow T^{o} = 25^{o}C$
$F \longrightarrow 1$ LB

Figure 14

COMPUTATIONAL SCHEME (Figure 15)

In the PSMS concept, major reliance is placed on computation for in-space dimensional determination and control.

COMPUTATIONAL SCHEME

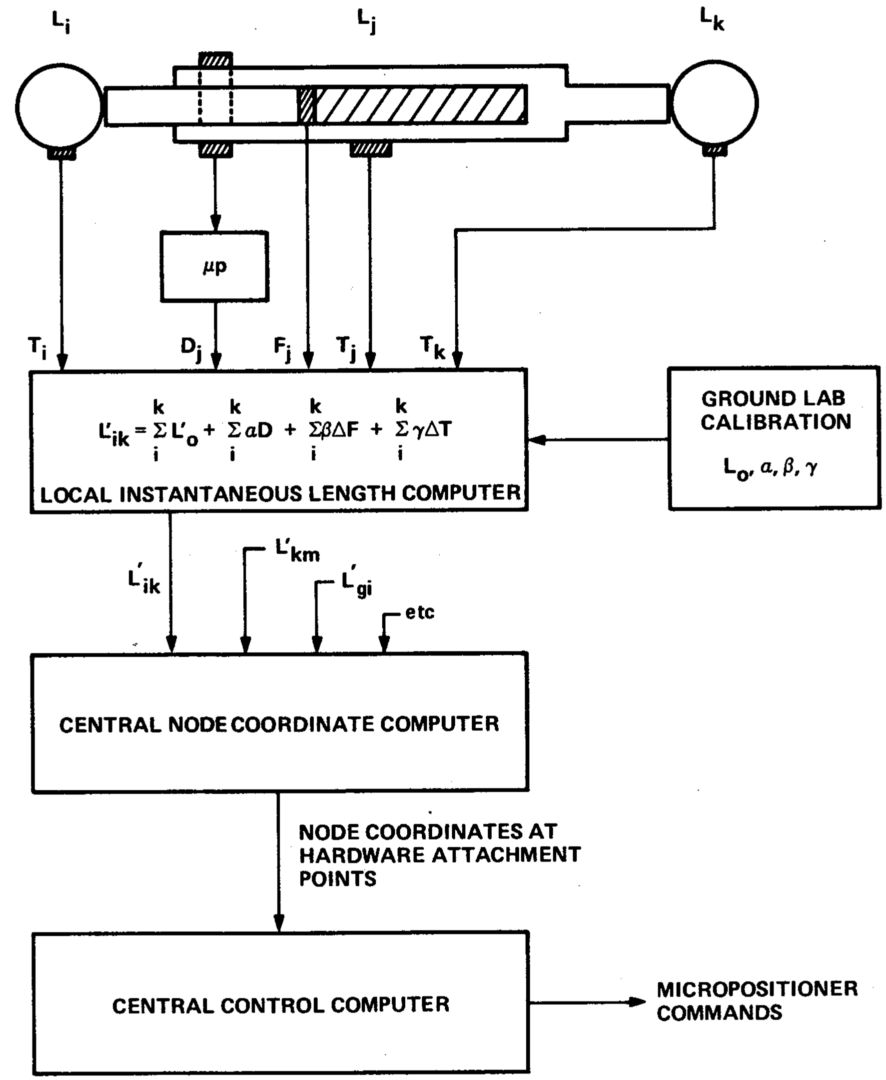
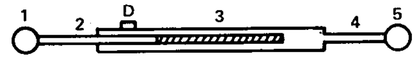


Figure 15

CUMULATIVE ERROR (Figure 16)

A preliminary analysis based on conservative estimates of elementary contributions indicates that the RMS error in attributed internodal length can be held to about 7 microinches for a 10-ft strut. A further analysis of error propagation in 3-dimensional networks (not described here) shows that, provided the figure is strong, RMS coordinate error will cumulate as the square root of the number of links traversed.

CUMULATIVE ERROR IN COMPUTED INTERNODAL LENGTH



$$\sigma_L^2 = \Sigma\, \sigma_{L_o}^2 + \Sigma\, (L\gamma)^2 \sigma_T^2 + \Sigma\, (L\beta)^2 \sigma_F^2 + \alpha^2 \sigma_D^2 + \Sigma\, (L\Delta T)^2 \sigma_\gamma^2 + \Sigma\, (L\Delta F)^2 \sigma_\beta^2 + (\Delta D)^2 \sigma_\alpha^2$$

	UNITS	1	2	D	3	4	5
L_T	IN	0.5	20		100		0.5
L_F	IN	0.5	20		-	20	0.5
A	IN^2	0.1	0.5		0.5	0.5	0.1
E	10^6PSI	30	10		10	10	30
γ_T	$10^{-6}/^oC$	10	0.5		0.5	0.5	10
ΔT	^oC	50	50		50	50	50
ΔF	#	1	1		0	1	1
ΔD	10^{-6}IN			10^5			
γ	$10^{-6}/^oC$	10	0.5		0.5	0.5	10
β	10^{-6}/#	0.3	0.2		0.2	0.2	0.3
α	IN/IN			1			
σ_T	^oC	0.1	0.1		0.1	0.1	0.1
σ_F	#	0.1	0.1		-	0.1	0.1
σ_D	10^{-6}IN			2			
σ_γ	$10^{-9}/^oC$	10	0.5		0.5	0.5	10
σ_β	10^{-9}/#	0.3	0.2		0.2	0.2	0.3
σ_α	10^{-6}IN/IN			20			
σ_{L,L_o}	10^{-6}IN	1		2			1
$\sigma_{L,T}$	10^{-6}IN	0.5	1		5		.5
$\sigma_{L,F}$	10^{-6}IN	-	0.4		-	0.4	-
$\sigma_{L,D}$	10^{-6}IN			2			
$\sigma_{L,\gamma}$	10^{-6}IN	.25	.5		2.5		.25
$\sigma_{L,\beta}$	10^{-6}IN	-	-		-	-	-
$\sigma_{L,\alpha}$	10^{-6}IN			2.0			

$\sigma_L = 6.9\mu IN$

Figure 16

THE PRECISION SELF-METERING STRUCTURE: SUMMARY

STRUCTURE

1. SHUTTLE TRANSPORTABLE
2. EVA OR MANIPULATOR ERECTABLE
3. GEOMETRICALLY VERSATILE
4. USABLE WITH OR WITHOUT METROLOGY
5. LOW PRECISION OF MANUFACTURE
6. WIDE CHOICE OF MATERIALS
7. DEMOUNTABLE AND REUSABLE

METROLOGY

1. PERMANENT, ONE-TIME GROUND CALIBRATION
2. NO SETUP, ALIGNMENT OR CALIBRATION IN SPACE
3. CALIBRATION UNAFFECTED BY STORAGE OR HANDLING
4. INHERENT LONG-TERM PHYSICAL STABILITY
5. NO MECHANICALLY ACTIVE OR DRIVEN PARTS
6. LOW POWER REQUIREMENTS
7. CONVENTIONAL MATERIALS AND COMPONENTS
8. RMS ERROR IN 10 FT LINK $\approx$ 7 MICROINCHES
9. RMS COORDINATE ERROR $\approx$ 7 MICROINCHES X $\sqrt{\text{NO. OF LINKS TRAVERSED}}$

Figure 17

COMMENTS OF GENERAL INTEREST FROM QUESTIONS AND ANSWERS

Precision Self Metering Structure

Ageing of Quartz

The long term dimensions and stability of quartz (the established need is 1 part in 10^9) requires further investigation before applications to precision systems. Present knowledge shows that aged quartz springs are more stable than new units. For this application, some insight into ageing effects are anticipated as a result from the first flight of the LDEF which anticipate carrying a simplified model of the Precision Self Metering Structure as one of the Experiments.

ELECTRICAL POWER LOSS FROM HIGH-VOLTAGE POWER CIRCUITS THROUGH PLASMA LEAKAGE

By Henry Oman, Boeing Aerospace Company

THE BOEING COMPANY
SEATTLE, WASHINGTON

SOLAR POWER SATELLITE (Figure 1)

The phenomenon of power leaking through plasma can best be explained by considering a solar power satellite which would have a large high-voltage solar-cell array.

A solar power satellite is the energy-converting portion of a system which injects into Earth-surface public utilities power generated from sunlight in a geosynchronous orbit, 35,693 km altitude. Other elements of the system are the microwave beam that transmits the power to Earth, and the receiving station with its array of antenna elements, rectifiers, and equipment for converting the collected power to high-voltage 60-Hz alternating current that is delivered to distributing utilities. A solar power satellite having solar cells in simple trough-type concentrators is shown in Figure 1, along with the other major system elements and their power losses.

To be practical, a solar power satellite must be large, in the order of 100 km^2. This is because the microwave beam should be sharp enough to focus its energy on a reasonable size receiving station, say 60 km^2 in area. An antenna generating such a beam would have to be about a kilometer in diameter. It can then transmit enough power to support on Earth a receiving station that delivers about 10 gigawatts (GW) of power. For reference, the larger nuclear power plants in operation in 1977 deliver about one GW.

Constructing a solar power satellite in low-Earth orbit, followed by self-powered electrically-thrusted transfer to its geosynchronous operating station seems to be the best approach. However, self-powered orbit transfer requires the generation of high voltage directly from the solar-cell array. About 1800 volts is the limit because at higher voltages current leaking through the plasma in low Earth orbit would steal too much power from the solar array.

In geosynchronous orbit, where there are only about 100 electrons per cm^3. the leakage current through the plasma will be insignificant, even when voltages up to 100 kV are generated in the solar array. However, ion engines used to control attitude and station location will generate charge-exchange plasma which can provide a path for leakage-current flow out of the solar cell array.

SOLAR POWER SATELLITE

CAN DELIVER 10 GW

ON EARTH

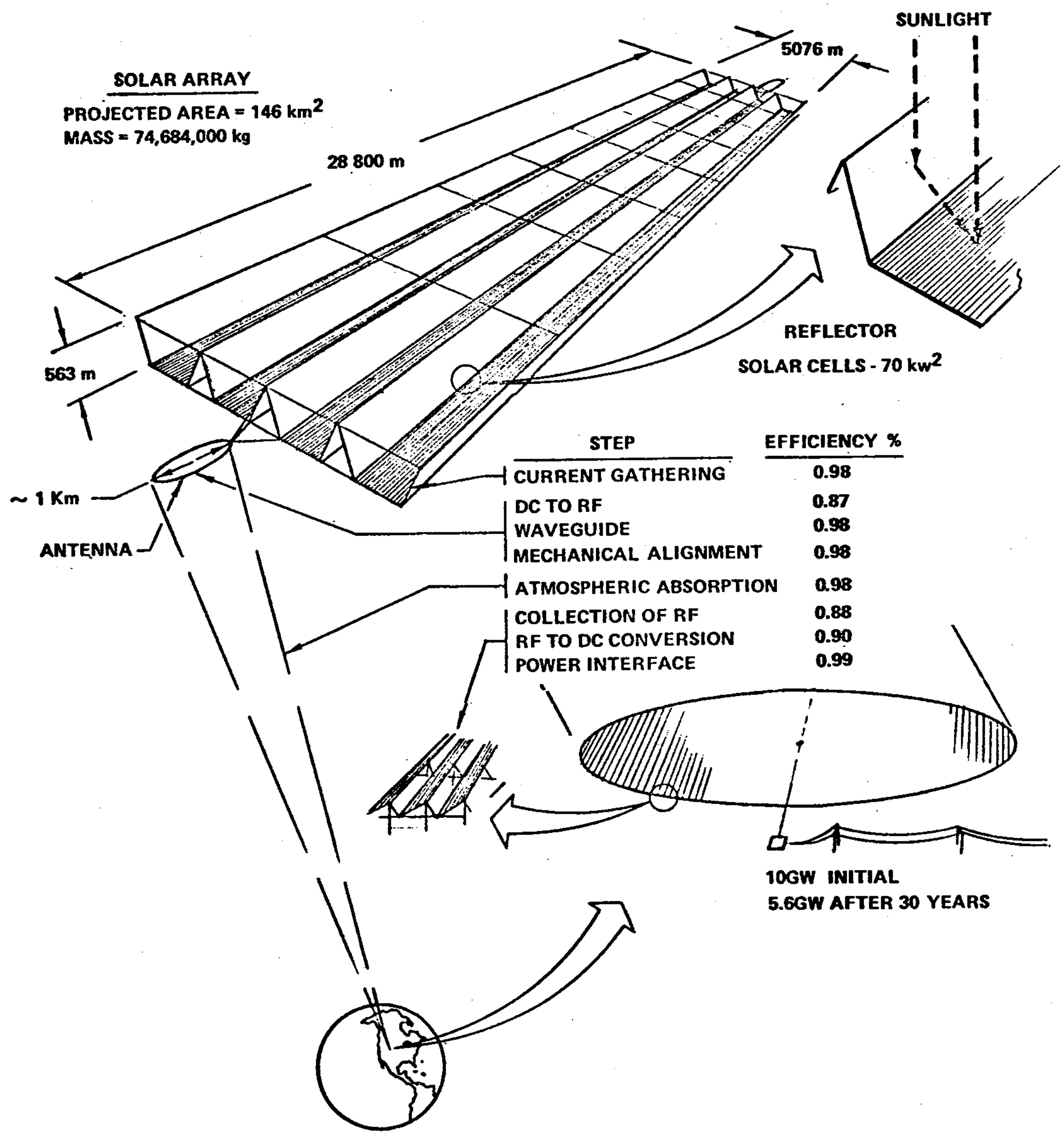


Figure 1

SPACE ENVIRONMENT (Figure 2)

The environment for the solar power satellite is summarized here, with altitude plotted in Earth radii, nautical miles, and kilometers horizontally on a logarithmic scale, but with the center of the Earth brought from minus infinity to the edge of the illustration.

The Earth's magnetic field does not directly affect the high voltage solar array of a solar power satellite, but it controls other phenomena that do affect the array. These phenomena vary by orders of magnitude in intensity as a result of solar activity induced changes in the Earth's magnetic field and particle arrival rates. For example, the ionospheric layers, of which the F_1 and F_2 are in the 100 nautical mile (NM) to synchronous-altitude operating regime, are affected not only by the magnetic fields, but also by the time of day and season of the year. During the day the ultraviolet in the sunlight ionizes the oxygen and nitrogen neutral atoms of the air, and produces over 10^6 electrons per cm^3. At night the recombination of electrons with ionized oxygen produces the air-glow. At synchronous altitude the normal electron count falls to 100 per cm^3, and 60 cubic kilometers must be swept to find a coulomb of charge.

PLASMA ENVIRONMENT VARIES WIDELY BETWEEN 500 AND 35693 KM

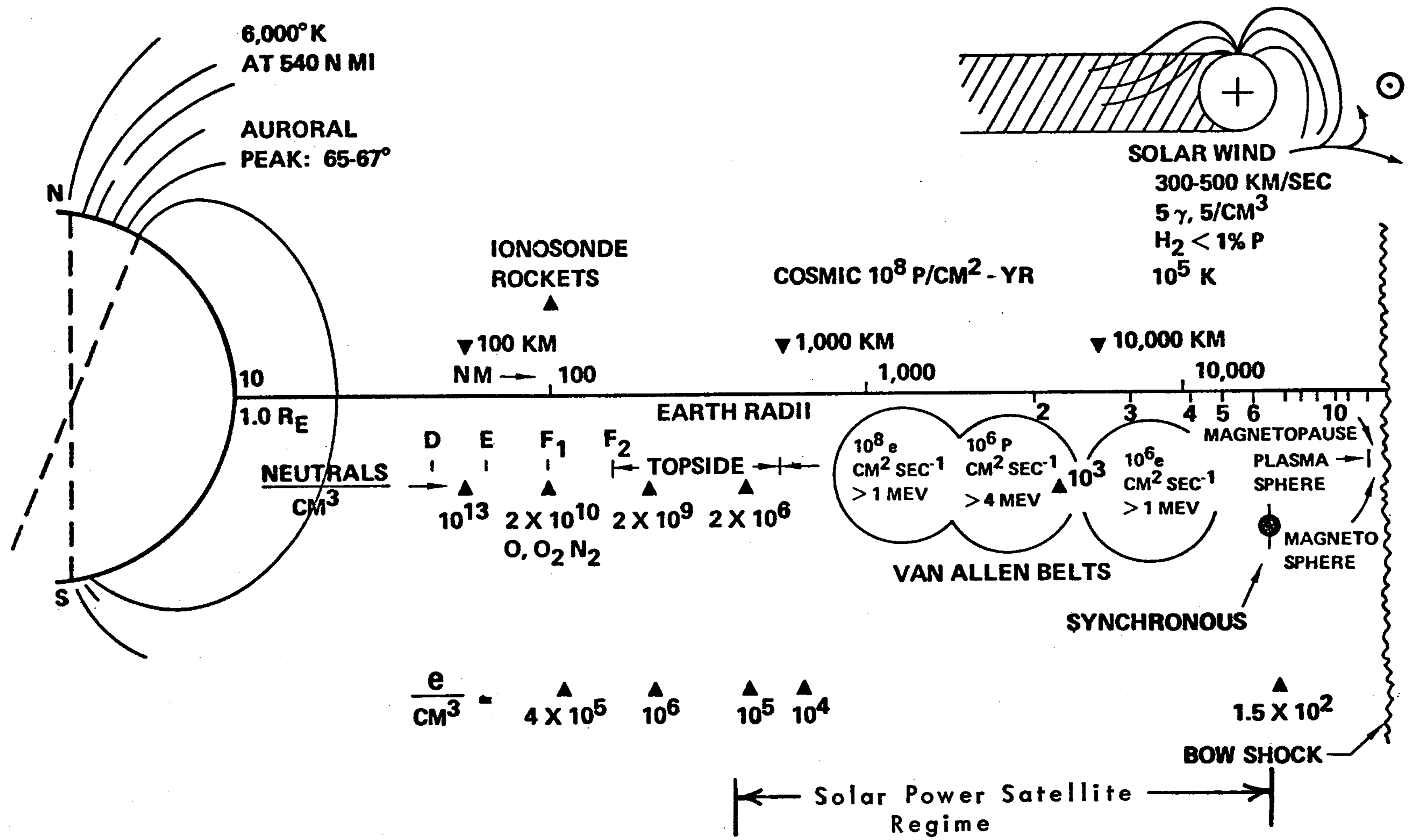


Figure 2

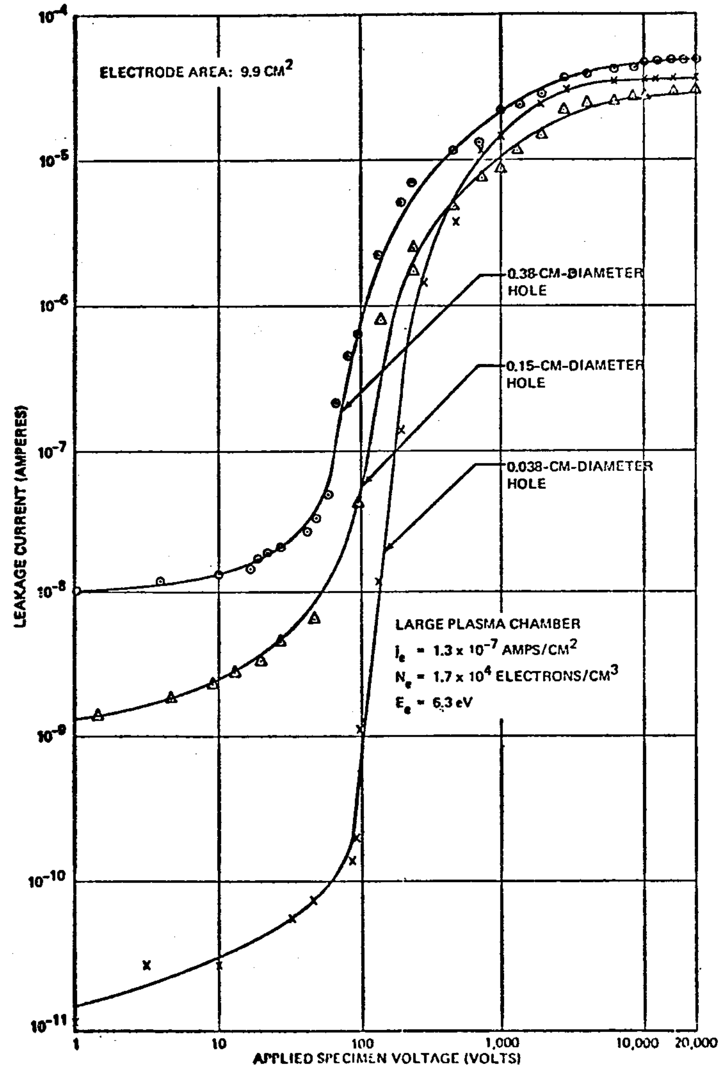
PINHOLE CURRENT COLLECTION (Figure 3)

Current flow through pinholes in insulation is an important mechanism in the escape of power from high-voltage solar arrays through a plasma path. Kennerud measured the current flowing from plasma through pinholes while varying pinhole size, insulation type, area of electrode and surrounding insulation, shape of pinhole, and type of insulation adhesive. Environmental and electrical parameters that were varied are plasma density, voltage level and polarity, length of plasma exposure, and background pressure.

In one of his tests he used 0.0127 cm Kapton insulation and measured current flow through the pinholes having hole diameters of 0.038 cm, 0.152 cm, and 0.381 cm. The Kapton was bonded to a stainless steel disc with conductive epoxy and mounted in a Teflon holder.

The measured leakage current with the electrode positive is shown here. At low voltages (<40 volts) the collected plasma current strongly depends upon hole size and is nearly independent of voltage. In this voltage range the collected plasma current density of the largest two holes is of the same order of magnitude as the random electron current density calculated from Langmuir probe data. The relatively low plasma current density collected by the 0.015-inch-diameter hole may be an insulation shielding effect; the hole diameter and depth, being nearly equal, make it a relatively "deep" hole whose walls limit the ability of the applied voltage to collect plasma electrons.

At intermediate voltages (+60 volts to +100 volts), the collected plasma currents start rising rapidly with increasing voltage. At high voltages (+1,000 volts to 20,000 volts) the currents for all three hole sizes are roughly equal and do not increase appreciably as voltage is increased.



At Above 200 Volts the Hole Size in 1.25 μm Kapton Ceases to Significantly Affect Leakage Current.

Figure 3

POWER LOSS BY LEAKAGE THROUGH PLASMA (Figure 4)

The space between 400 km altitude and the orbits of geosynchronous satellites contains neutral atoms, free electrons, positive ions, and high-energy charged particles. The high-energy particles, although damaging to solar cells and optical surfaces, are not numerous enough to carry a significant current. The free electrons, generated each morning when ultraviolet photons ionize neutral atoms, have energies of around one to two electron volts. This energy is dissipated in reactions with neutral atoms and ions increasing the temperature of the medium to the region of 500K to 2000K.

A positively charged spherical electrode, say one cm in diameter, will collect electrons when inserted into a plasma. The volume in which electrons are influenced by the electrode, called a sheath, is much larger than the sphere. Some of the electrons will orbit around the electrode and escape back out of the sheath. Current collection is then said to be orbit-limited and is affected in a complex manner by the radius of the electrode, the voltage of the electrode, and the temperature and density of the free electrons.

The high-voltage solar-cell array for a solar power satellite looks more like a sheet electrode than like a spherical probe. For example, let us assume that 10 km^2 of a solar power satellite array is deployed to supply 1500-volt power for electric propulsion thrusters for raising the satellite from low-Earth orbit, say 500 km, to geosynchronous orbit. We determined that at 500 km the electron-sheath extends to a few meters above the plane of the solar cells, in the range of electron concentrations, electron temperatures, and array voltages of interest. The calculation of leakage current then simplifies into analyzing the rate at which electrons drift into an electron sheath having essentially the same area as the solar array.

The calculated leakage currents from a 1500-volt array for several altitudes are shown in the table.

LEAKAGE CURRENT FROM POSITIVELY CHARGED SOLAR ARRAY

Array Altitude, Km	Electron Density, N_e Electrons/cm^3	Electron Temperature ^oK	Leakage Current		Power Loss, Percent of Generated
			nA/cm^2	Amperes per 1500 V String*	
500	6×10^5	3,000	824.5	0.8494	7.72
700	2×10^5	3,000	274.8	0.2831	2.57
1,000	7×10^4	3,000	96.19	0.0990	0.90
2,000	2×10^4	3,200	28.38	0.0292	0.265
5,000	1×10^4	4,400	16.64	0.0171	0.156
10,000	8×10^3	5,400	14.75	0.0152	0.138
20,000	2×10^3	9,000	4.76	0.0049	0.044
30,000	1×10^2	13,600	0.29	0.0003	0

* The string is 0.404 m by 255 m, with an area of 133.02 m^2

Figure 4

SOLAR ARRAY POWER LOSS THROUGH CHARGE-EXCHANGE PLASMA (Figure 5)

Charge-exchange plasma is generated in the downstream of an electric thruster when the beam ions interact with neutral atoms escaping from the thruster. The charge exchange plasma was first discovered to be a conducting medium in the path of electrons going from the thruster to the solar array during tests of the ATS-6 spacecraft in a large vacuum chamber. The spacecraft was biased at +15 volts relative to the thruster neutralizer, and substantial electron currents were observed flowing to the spacecraft.

The ions in the charge exchange plasma will be propelled by electric fields away from the ion beam and toward the back of the thruster. The positive ions in this plasma, not being readily absorbed by the solar array, constitute a minor part of the leakage current. The electrons, on the other hand, can funnel through holes in any solar array insulating surface and can rob significant power from the solar array.

CHARGE EXCHANGE PLASMA IS GENERATED WHEN BEAM IONS IONIZE NEUTRAL PROPELLANT ATOMS DRIFTING OUT OF THRUSTER

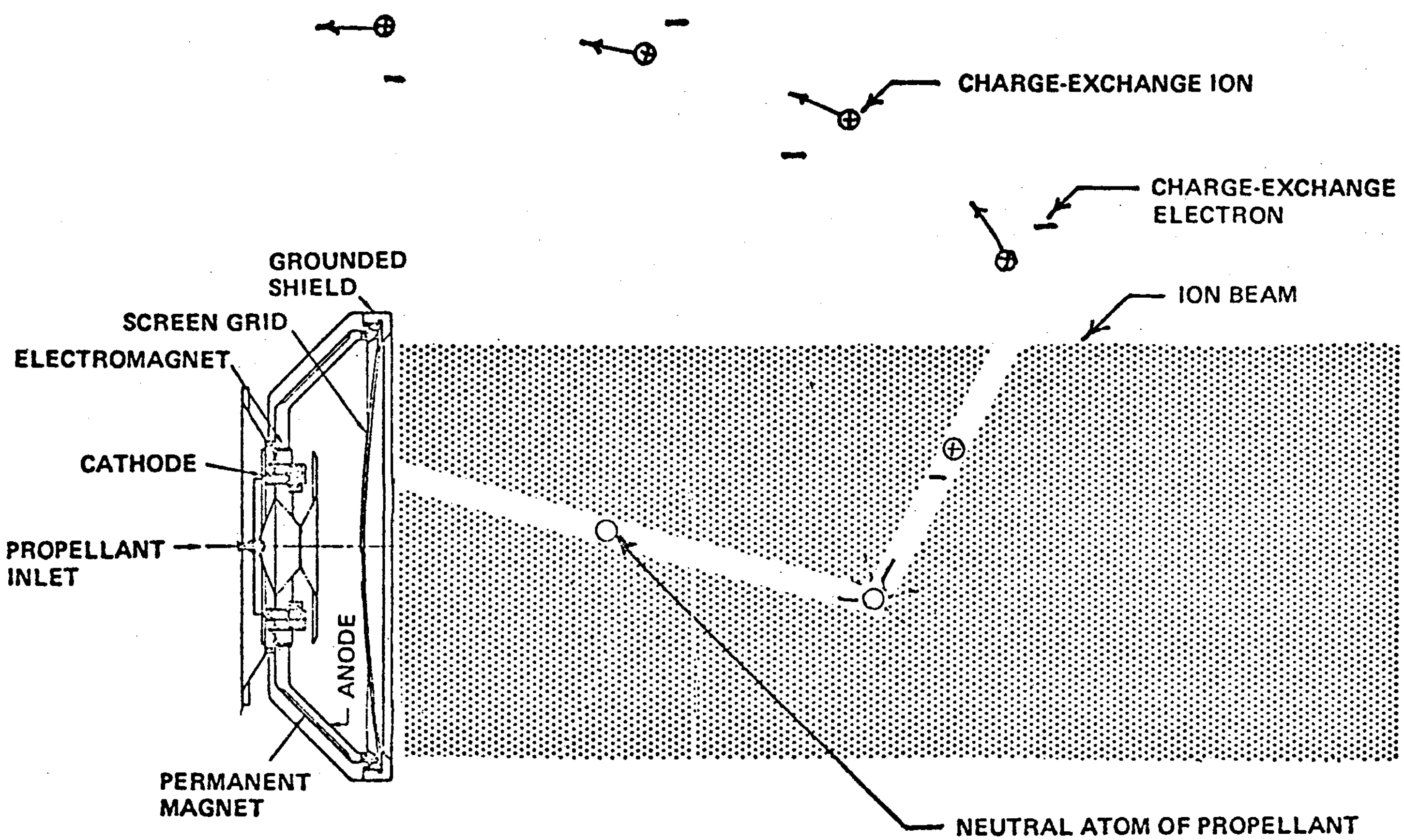


Figure 5

LIMITING POWER LOSS THROUGH CHARGE-EXCHANGE PLASMA (Figure 6)

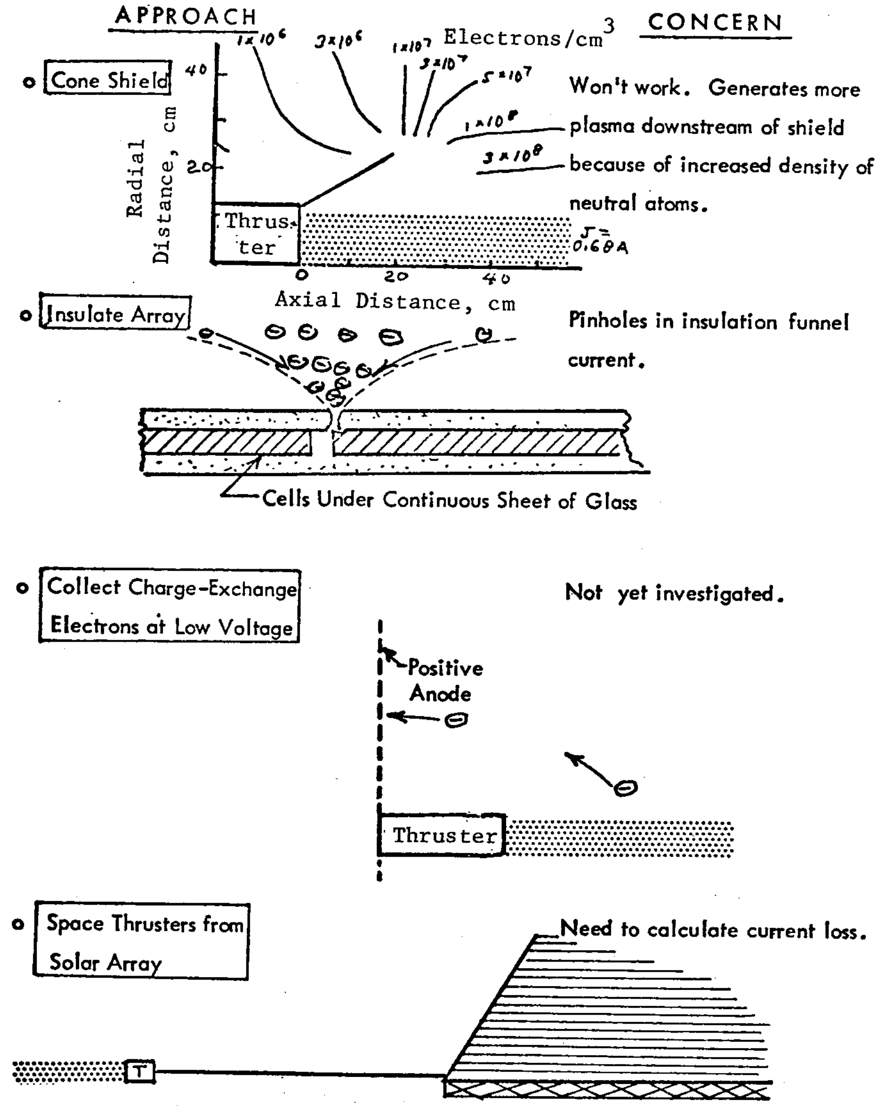
Methods of avoiding the solar array loss in output caused by charge-exchange plasma conduction are shown. Using the cone shield does not reduce losses. The cone does move the apparent source of plasma generation downstream and further from the solar array. However, it also increases the density of the neutral atoms, and the net result is an increase in the leakage current from the solar array.

Insulating the solar array would eliminate the leakage-current loss were it not for the pinholes. Kennerud showed that within the plasma sheath surrounding the solar array, electrons will funnel into a pinhole from a large volume of the plasma. If the electron current is great enough, the pinholes will enlarge as the surrounding insulation sublimates away.

A third method of controlling leakage current going through the charge-exchange plasma is to collect the electrons with an anode before they can get into the solar array. Kaufman dismisses this approach with the note that it consumes too much power. However, most of the electrons can be collected on a 20-volt anode. Using Kaufman's generation rate of charge-exchange plasma gives a current of 56.4 kA, which at 20 volts represents 1.13 MW or 2.2 percent of the 52 MW consumed by the 800 thrusters.

If the charge-exchange plasma generation is limited by the supply of neutral argon atoms released by the thrusters, then the collecting plate will carry only 7.75 kA, which at 20 volts represents 155 kW or 0.3 percent of the power supplied to the thrusters. The anode could be a thin sheet of metal connected to the thruster structure through a 20-volt power supply.

If nothing else works, then the ion thrusters could be spaced away from the solar array.



Ways of Reducing Power Loss Through Charge-Exchange Plasma.

Figure 6

CONCLUSIONS

Generation of power at high voltage, around 40 kV, is advantageous on a solar power satellite. For example, power for the rf amplifiers would be carried by buses from solar cells as far as 10 km away, and even at 40 kV the current at the rotary joint between the array and antenna will be around 200 kA. Bus weight is reduced as voltage is raised. Also, generating power at 40 kV, a good input voltage for high-power klystron rf power amplifiers, avoids the need for heavy power conversion equipment.

Leakage of current through plasma can constitute a significant power loss from a high-voltage solar array. For example, at 300 km altitude a 2 kV array can barely generate enough power to feed the plasma losses. We believe that 477 km is a good altitude for assembling the solar power satellites, and that supplying power at 1.8 kV to the thrusters for orbit transfer is appropriate. Even at this voltage the I^2R losses in the power buses will be significant, but we need only about one-fourth of the satellite solar array for powering the thrusters.

In geosynchronous orbit where the satellite generates power and delivers it to Earth with a microwave beam, and the electron density is only around 100 per cm^3, plasma leakage current will be trivial if ion thrusters are not operating.

Leakage current through charge-exchange plasma, which is generated whenever ion engines are operated for orbit transfer or station keeping, is a phenomenon that is real but not yet fully understood. Leakage current through charge-exchange plasma might be reduced by shields, spacing, or ion collection at low voltage, but much more work needs to be done before we can be sure.

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16. Abstract

These proceedings include a compilation of the papers presented during a seminar which provided Government and Industry representatives an opportunity to exchange information, to review the present status of related technology, and to plan the development of new technology for Large Space Systems. These papers were divided generally into two major areas of interest: the first group addressed subjects pertinent to large antenna systems; the second group addressed technology related to large space platform systems.

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